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ABSTRACT

Presents results of study to determine the degree to which the Lunar Orbiter (in its present or in a modified form) is capable of supporting other scientific experiments. The study shows that other experiments can be accommodated by minor hardware changes because of the operational flexibility available in the present design.

KEY WORDS

Lunar

Experiments

Orbital

Spacecraft

Adaptability

Scientific investigation

Lunar Orbiter

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1.0 INTRODUCTION

1.1 PURPOSE

This document, D2-100369-1, is the first of two volumes of the final written report on "A Research Study of the Lunar Orbiter Spacecraft Regarding Its Adaptability to Other Scientific Investigations", performed under Contract NAS 1-4959, dated April 28, 1965. The second volume, D2-100369-2, provides background description of the Lunar Orbiter appropriate to the adaptability study.

1.2 SCOPE

The study examined the adaptability of four distinct configurations to ten designated scientific experiments. The four configurations, referred to herein as Cases I, II, III and IV and the ten experiments were specified in the contract Statement of Work L-5382. The configuration cases are described in Table 1.2-1 for convenience.

TABLE 1.2-1

CONFIGURATION CASES

Case I - Complete present Photo Subsystem. Allowable weight, 920 pounds.

Case II - High Resolution Photo Capability removed. Allowable weight, 860 pounds.

Case III - Medium Resolution Photo Capability removed. Allowable weight, 860 pounds.

1.2 (Continued)

Case IV - Photo Capability totally removed. Allowable weight, 860 pounds.

All Cases: Existing micrometeoroid and radiation dosimeters removed. Shroud configuration for Block I to be retained.

Launch vehicle for Block I to be used.

The ten scientific experiments are described in Appendices A and B herewith, which are reproductions from the contract Statement of Work. It will be noted that the experiments may be classified as follows:

Surface Related	Space Related
Gamma Radiation	Micrometeoroid
Infrared	Solar Plasma
BiStatic Radar	Magnetic Field
Photometry/Colorimetry	Selenodesy
X-Ray Fluorescence	
Radiometer	

2.0 SUMMARY

2.1 METHODOLOGY

The experiment description data as provided by the Statement of Work was supplemented by a literature search wherever necessary

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2.1 (Continued)

for purposes of configuration definition and subsystem modification studies. The results of the search are summarized in Appendix C. It is to be noted that wherever a discrepancy between the Statement of Work and the literature search existed, the study uses the data supplied by the Statement of Work since the literature search was limited to state-of-the-art equipment.

The general system and subsystem operational flexibility and constraints were reviewed and a range of feasible modifications, varying in complexity and providing a wide range of capability of supporting combinations of scientific experiments, was identified.

The general parametric study referred to above was used to establish ground rules for integration of scientific experiment groupings into the four configuration cases. The experiment operational requirements, and the photo subsystem modifications as specified by the Statement of Work were used to define mission parameters and operational sequences.

System configuration layouts were generated for several experiment groupings in order to illustrate the capability of mechanical integration and to provide baseline data for subsystem studies.

Subsystems studies relating to two specific configurations and mission profiles and sequences were performed in order to extend the parametric data to specific cases and to provide a realistic

2.1 (Continued)

identification of subsystem modification requirements involving experiment groups.

2.2 CONCLUSIONS

Each of the four Lunar Orbiter spacecraft configuration cases stipulated in the Statement of Work can accommodate combinations of scientific experiments. Regardless of the configuration chosen, changes to the vehicle and its subsystems will be limited to adaptations and rearrangements made necessary by the requirements of the experiment instrumentation. The performance of the spacecraft will be retained in all important respects.

Optimization of the spacecraft/experiments should be accomplished through the design of appropriate instrumentation and by utilization of the operational flexibility of the vehicle. The capability of the Lunar Orbiter to select from a wide range of orbit geometrics, is of significant importance. Extended life in orbit and the ability, under certain conditions, to made orbital changes may be of equal or greater value in certain missions.

The planning for space and for surface missions and for combinations of both must recognize that the spacecraft is a space-stabilized vehicle which relies on celestial reference and solar power. It is, therefore, space-oriented during the major portion of each orbit. Surface-related experiments can be accommodated, however, by orienting the spacecraft to the lunar surface for a

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2.2 (Continued)

portion of each orbit in the manner employed for the present photographic subsystem. Extended surface area coverage can be achieved by repetition of the experiment on several orbital passes. Space-experiments may be performed continuously, if desired.

Mounting of additional sensors may introduce a requirement for a general rearrangement of components, including the photographic subsystem, on the equipment mounting deck in order to preserve spacecraft balance.

Power requirements of the experiments can be met without subsystem modification. This can be accomplished by a mission profile design providing an appropriate time balance between operation on solar and battery power.

Communication subsystem and data storage requirements associated with additional experiments can be accommodated up to a rate of 100,000 bits/second in intermittent experiment operation, by an addition of a tape recorder. The accumulated data store will be transmitted in a compressed time period over the present video link on a time share basis, with video data, if necessary.

Attitude control subsystem requirements associated with experiment mounting and experiment orientation requirements can be met with minimal modifications. Increased reaction control nitrogen gas requirements, associated with increased number of

2.2 (Continued)

spacecraft orientation maneuvers, can be met by an addition of manifolded nitrogen tanks. The problems of increased moments of inertia, associated with increased spacecraft weight and/or boom deployment, can be resolved either by increasing nitrogen tank capacity and thruster size or by reducing spacecraft maneuver rates. The latter would involve a minor modification of the closed loop electronics only.

Additional control functions, associated with experiment control can be met by modification of the programmer output matrix and provision of additional switching functions.

Photographic subsystem modifications, involving deletions of either the high resolution or medium resolution portion of the subsystem, offer limited advantage in terms of volume and weight available for experiments. The volume accrued is neither required nor recommended for use in experiment accommodation. The deletion of the high resolution portion of the subsystem appears to be attractive on the grounds of weight increment made available for alternate experiments and increased area coverage capability (fourfold). The deletion of the medium resolution portion of the subsystem does not appear to be justified from either the weight or area coverage increment made available.

The additional experiment payload capability provided by configuration definitions of Cases I through IV would be 70, 38, 13, and

2.2 (Continued)

159 pounds respectively. This capability can be increased by careful mission profile design, involving propellant off-loading, up to 25 or 30 pounds without a significant effect on experiment performance capability. The total payload capability for Cases I through IV would thereby be increased to up to 100, 68, 43 and 189 pounds respectively.

A growth potential in excess of the above exists without major spacecraft modification by taking advantage of the modular construction of the spacecraft and availability of space qualified propellant tanks. This potential, resulting in an increase of payload capability up to 400 pounds and/or increase in mission range capability, can be realized contingent on upgrading the earth boost system to a level predicted for the SLV-3X system.

In summary, it is concluded that the Lunar Orbiter has a significant potential as a vehicle for scientific investigation of the lunar surface and lunar environment in a largely unmodified version. A potential for an additional capability increment exists and may be exploited in planetary as well as lunar exploration.

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3.0 PARAMETRIC FLIXIBILITY STUDIES

This section outlines the general system flixibility and constraints, and discusses the range of subsystem modifications in relation to their complexity, performance enhancement and adaptability to scientific experiments.

In particular; the orbital mechanics of translunar transit and lunar orbits, the geometry of providing area coverage under proper illumination and altitude conditions, are reviewed in order to establish a baseline for correlation of subsystem(s) and experiment(s) operational requirements and constraints. The available flexibility of individual subsystems is then discussed in the above context.

3.1 ORBITAL REQUIREMENTS IMPOSED BY EXPERIMENTS

An experiment generally defines certain orbital requirements in terms of:

a. Selenodetic location of the experimental area;

b. Dimensions of the area of coverage;

c. Solar illumination requirements;

d. Altitude range at which the experiment should be performed.

The effects of these requirements on mission profile design and the resulting subsystem requirements are discussed in the following subsections.

3.1.1 Location of Area of Interest

The location of the line of nodes, or the intersection of the lunar approach plane with any constant latitude plane other than

3.1.1 (Continued)

the equatorial plane, is primarily determined by the approach energy, time of launch and the inclination of the lunar approach hyperbola. The energy at arrival in conjunction with time of launch determine the direction from which the vehicle approaches the moon. Typical variation of the longitude of the approach direction as a function of transit time is shown in Figure 3.1.1.1. The latitude of the approach may vary between the approximate limits of $\pm 15^{\circ}$.

The perilune of the approach hyperbola is, to a first approximation, at a constant central angle from the approach direction, for a given approach energy, and the inclination of the approach hyperbola can be arbitrarily controlled either at launch or at the time of midcourse correction with a minor delta-velocity expenditure. As a result of the above the approach geometry is as shown for an arbitrary case in Figure 3.1.1.2, where the locus of the possible approach hyperbola perilune positions is obtained by a rotation of the given hyperbola about the approach direction.

Since the minimum delta-velocity expenditure for injection into a lunar orbit occurs at the perilune of the approach hyperbola where the velocity increment requirements are a shown in Figure 3.1.1.3, the optimum mission profile from the viewpoint of delta-velocity minimization would place the elliptical orbit perilune at the locus of the approach hyperbola perilunes and coplanar with the approach hyperbola. This combination is not generally possible.

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LONGITUDE OF LUNAR APPROACH

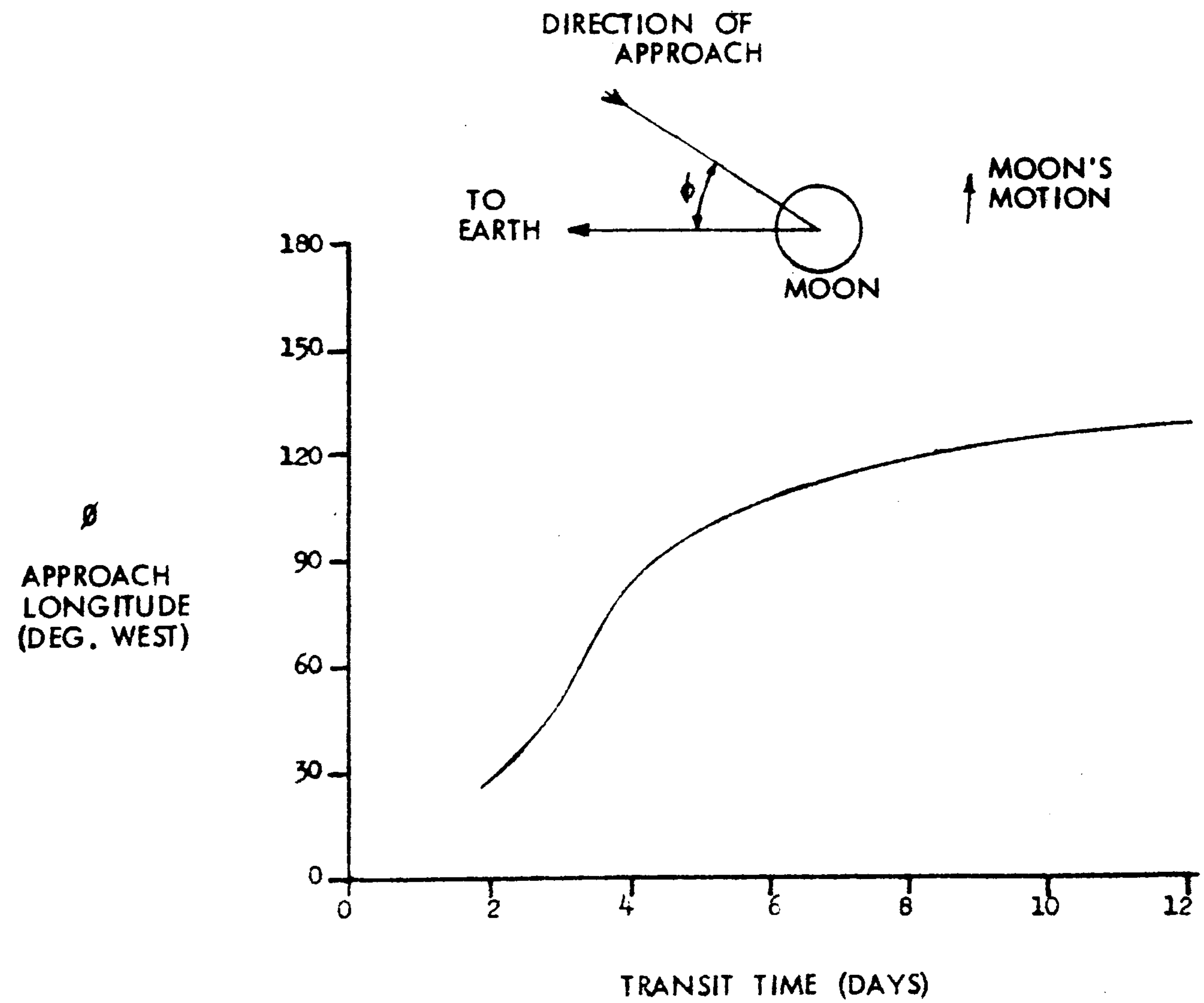

FIGURE 3.1.1.1

APPROACH GEOMETRY

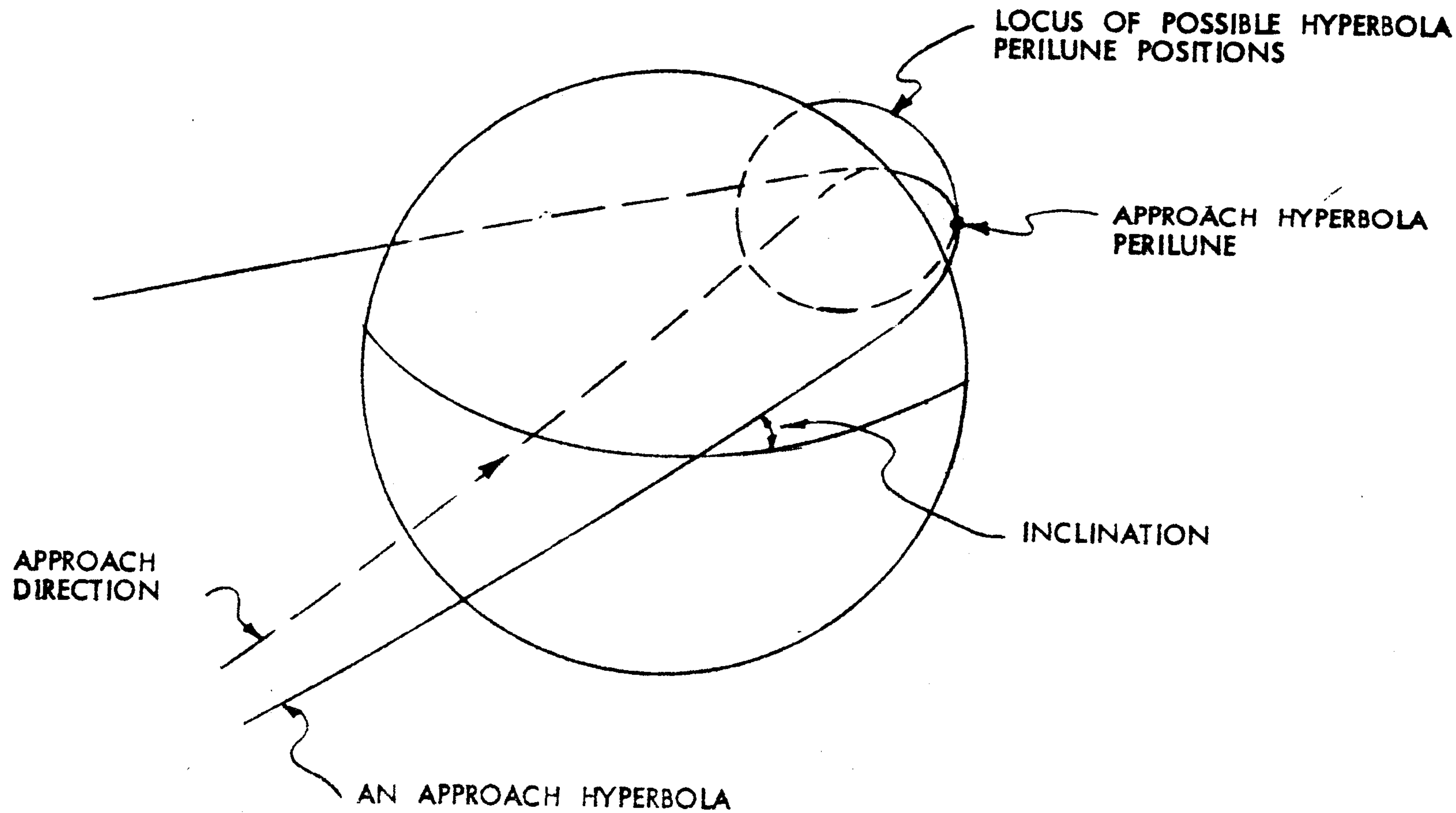

FIGURE 3.1.1.2

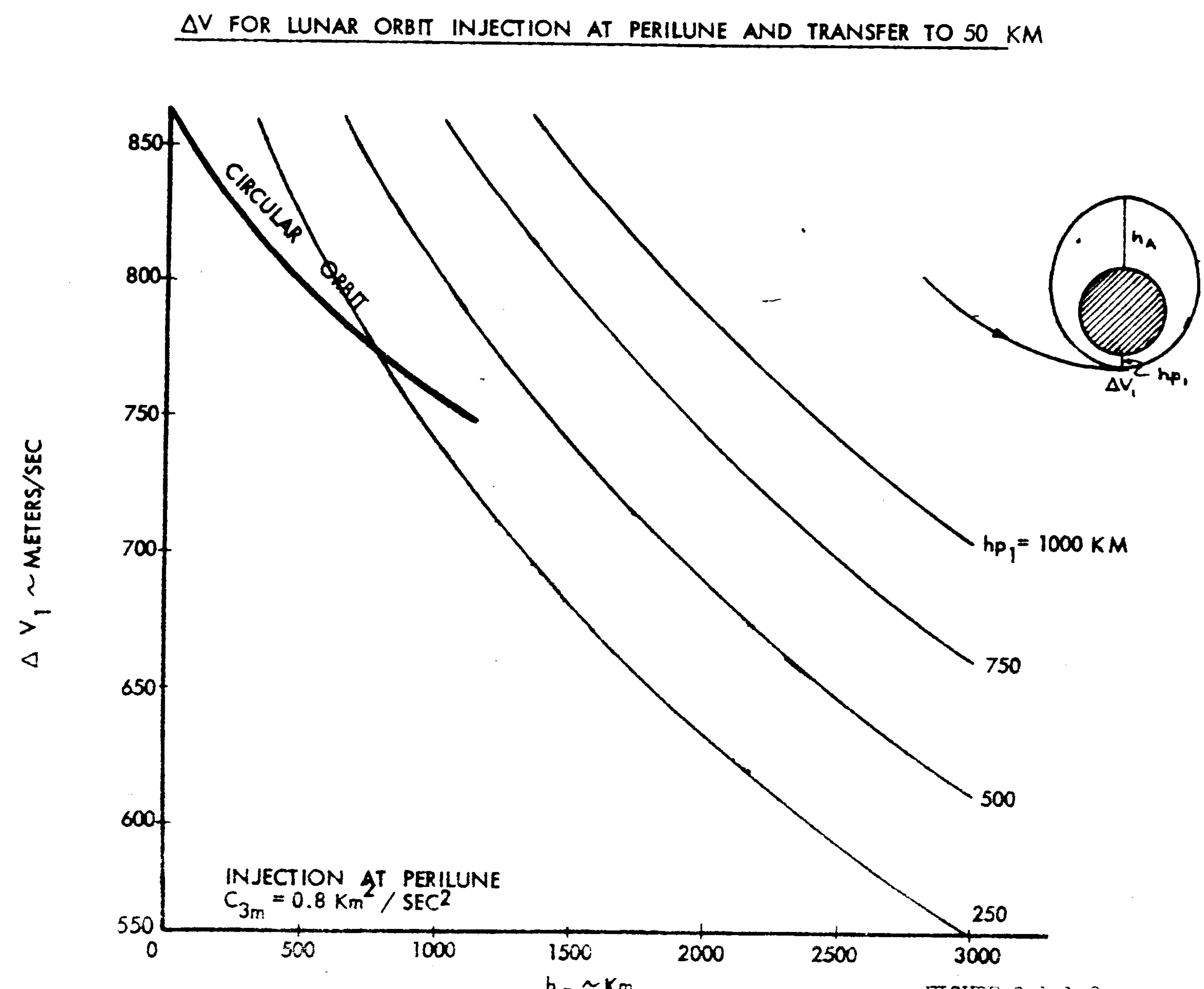
ΔV FOR LUNAR ORBIT INJECTION AT PERILUNE AND TRANSFER TO 50 KM
h_A
h_{p_1}
ΔV_1
h_{p_1} = 1000 KM
750
500
250
CIRCULAR ORBIT
INJECTION AT PERILUNE
C_{3m} = 0.8 Km^2/SEC^2
ΔV_1 ~ METERS/SEC
850
800
750
700
650
600
550
0
500
1000
1500
2000
2500
3000
h_a ~ Km

FIGURE 3.1.1.3

3.1.1 (Continued)

This definition of the mission profiles would, however, result in the following general features, with respect to target areas other than targets of opportunity.

a. The perilune of the elliptical orbit would not generally coincide with the experiment target latitude and, therefore, the experiment would be performed at a higher altitude than desirable. This is illustrated in planar projection in Figure 3.1.1.4.

b. The time of arrival at the target would be completely predetermined by the approach geometry and the location of the experimental target. The arrival time would follow the relation:

$$T = \frac{W_T - W_H}{\dot{W}_m + \dot{W}_o}$$

where

T = Time after injection to reach experiment location

W_T = Longitude of the target

W_H = Longitude of intersection of the target latitude plane with the approach hyperbola plane.

$\dot{W}_m$ = Rate of rotation of the moon.

$\dot{W}_o$ = Rate of precession of the lunar orbit.

The above relation is illustrated in Figure 3.1.1.5.

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EXPERIMENTAL TARGET AREA VS. PERILUNE LOCATION
MINIMUM INJECTION VELOCITY
PLANAR PROJECTION

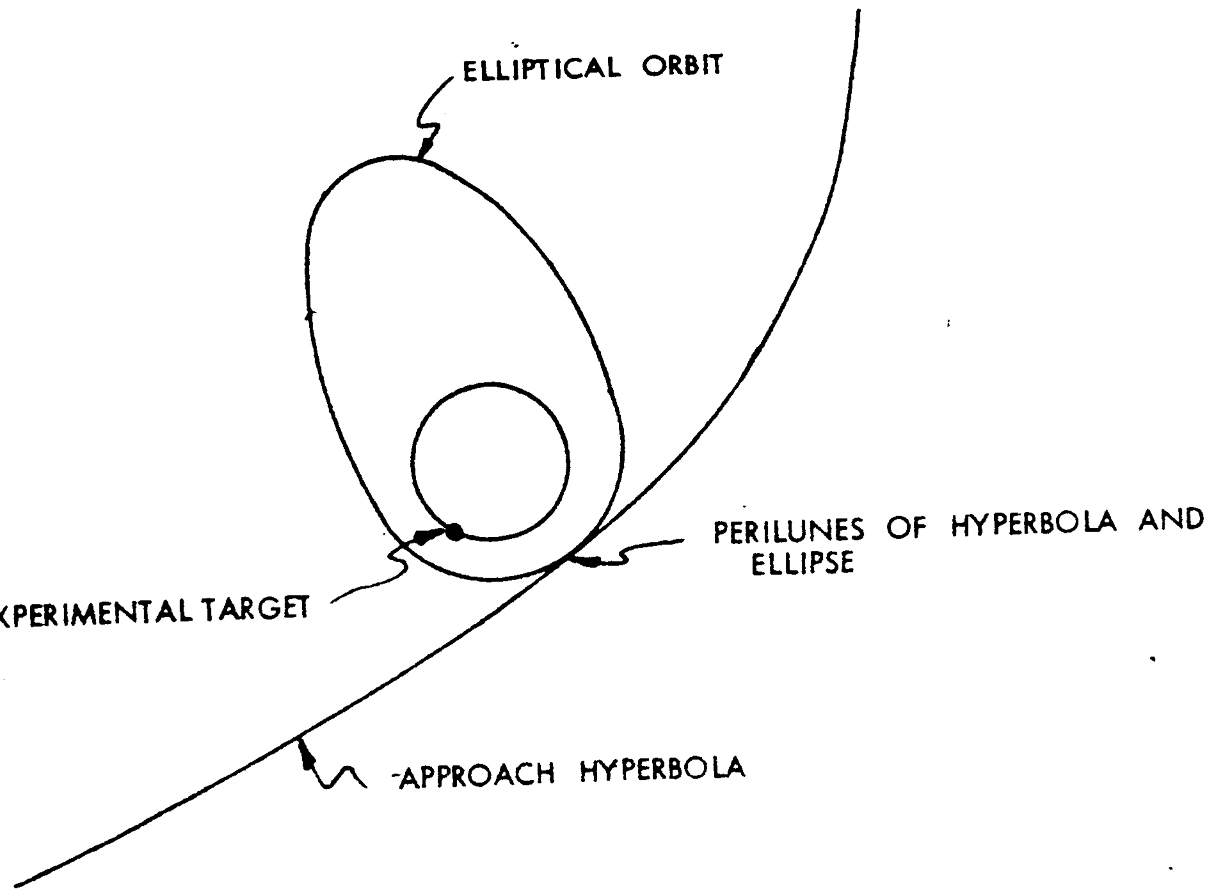
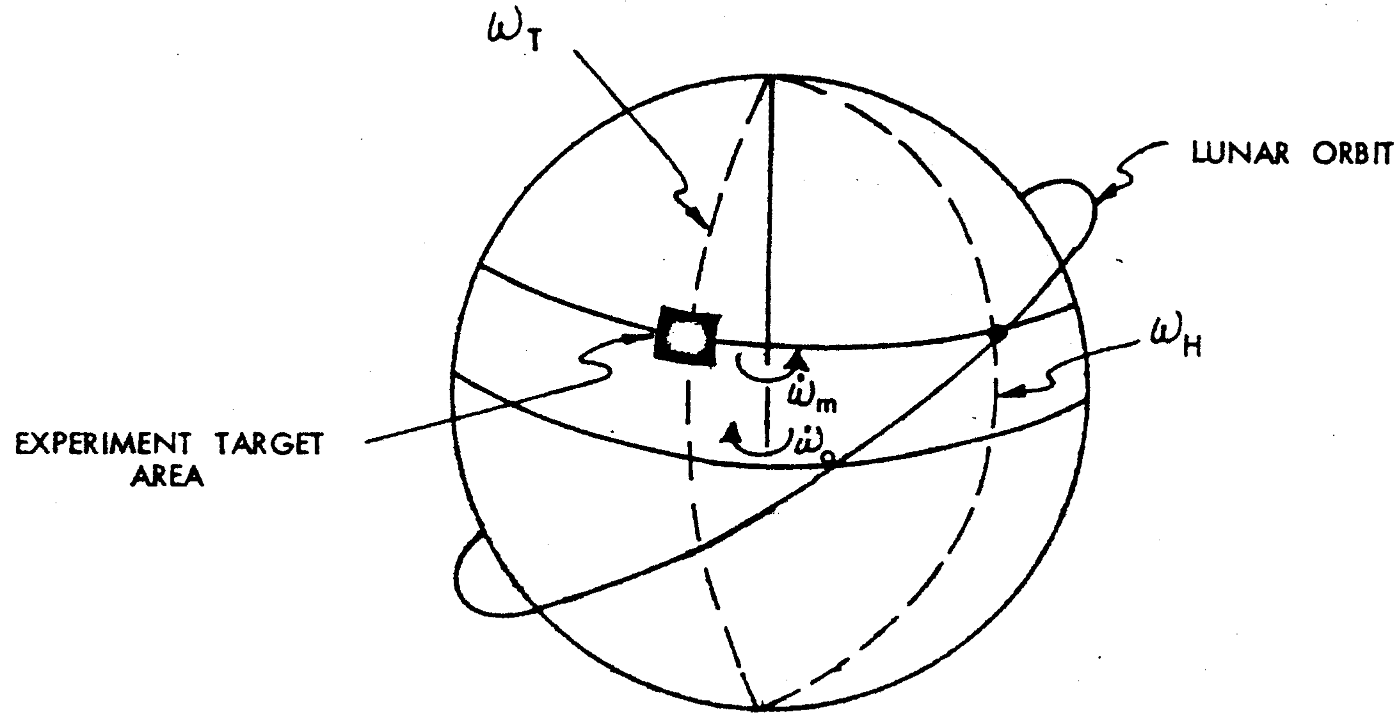


FIGURE 3.1.?.4

WAITING TIME TO REACH TARGET AREA



$$T = \frac{\omega_T - \omega_H}{\dot{\omega}_m + \dot{\omega}_o}$$

FIGURE 3.1.1.5

3.1.1 (Continued)

Additional delta-velocity relative to Figure 3.1.1.3 expenditures are required to achieve a flexibility of location of perilune at a latitude corresponding to the experimental target area latitude. Typical additional velocity increment requirements associated with this rotation of perilune (line of apsides rotation) are shown in Figure 3.1.1.6 with the orbital period as a parameter.

If positive control over the arrival time at the target is desired, in addition to the capability of controlling the latitude of lunar orbit perilune, it becomes necessary to provide the capability for an orbit plane change at injection. This mode of operation, is illustrated in Figures 3.1.1.7 and 3.1.1.8. Delta-velocity requirements for injection into the final orbit are shown in Figure 3.1.1.9 and 3.1.1.10 for launches in June and December of 1966 for a range of target longitudes of $\pm 60^\circ$ and target latitudes of $\pm 10^\circ$. This data is included for illustrative purposes only and is directly applicable only when a photographic mission in the near equatorial Apollo mission band of interest is concerned. The latter is particularly true since the launch dates include consideration of the photographic subsystem constraint of solar illumination at perilune of 60° and, therefore, includes the delta-velocity requirement associated with this constraint. Indirectly, the data of Figures 3.1.1.9 and 3.1.1.10 illustrates the trend in velocity expenditures incurred by providing increased operational flexibility relative

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VELOCITY REQUIREMENT VS. PERIAPSIS ROTATION AND ORBIT PERIOD

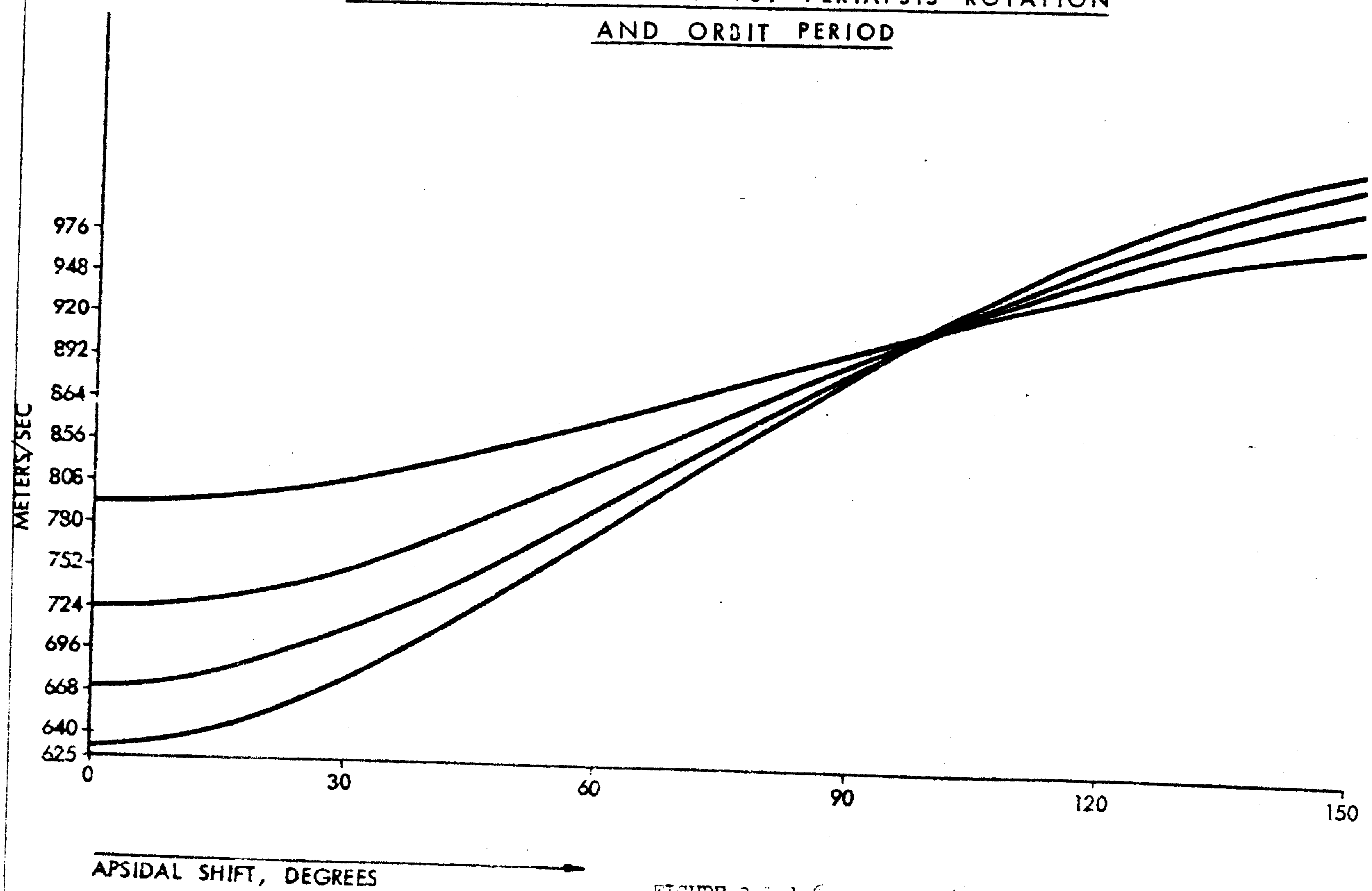

FIGURE 3.1.1.6

INJECTION MANEUVER

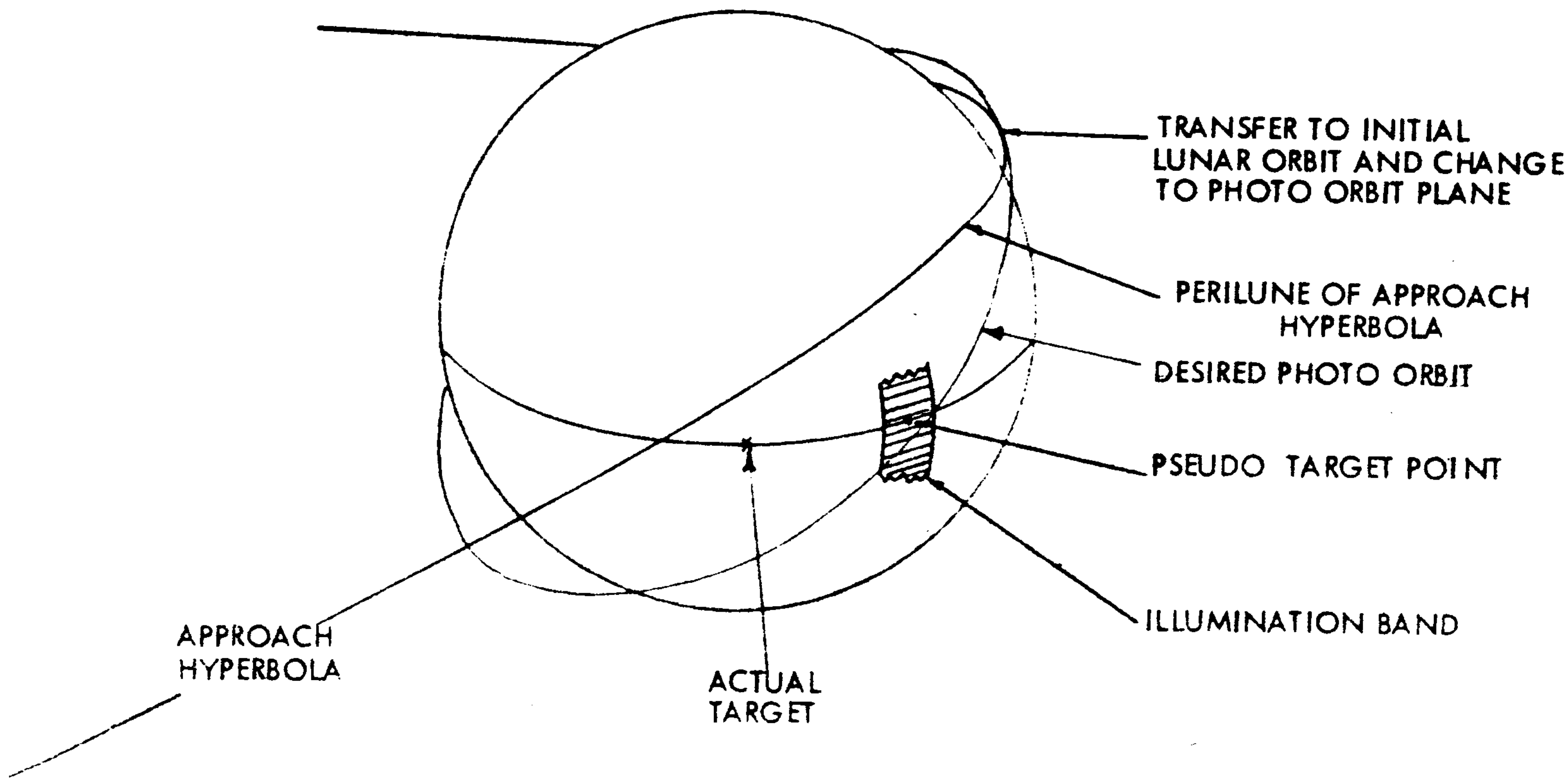

FIGURE 3.1.1.7

APPROACH ELLIPSE GEOMETRY
S-110 TRAJECTORY

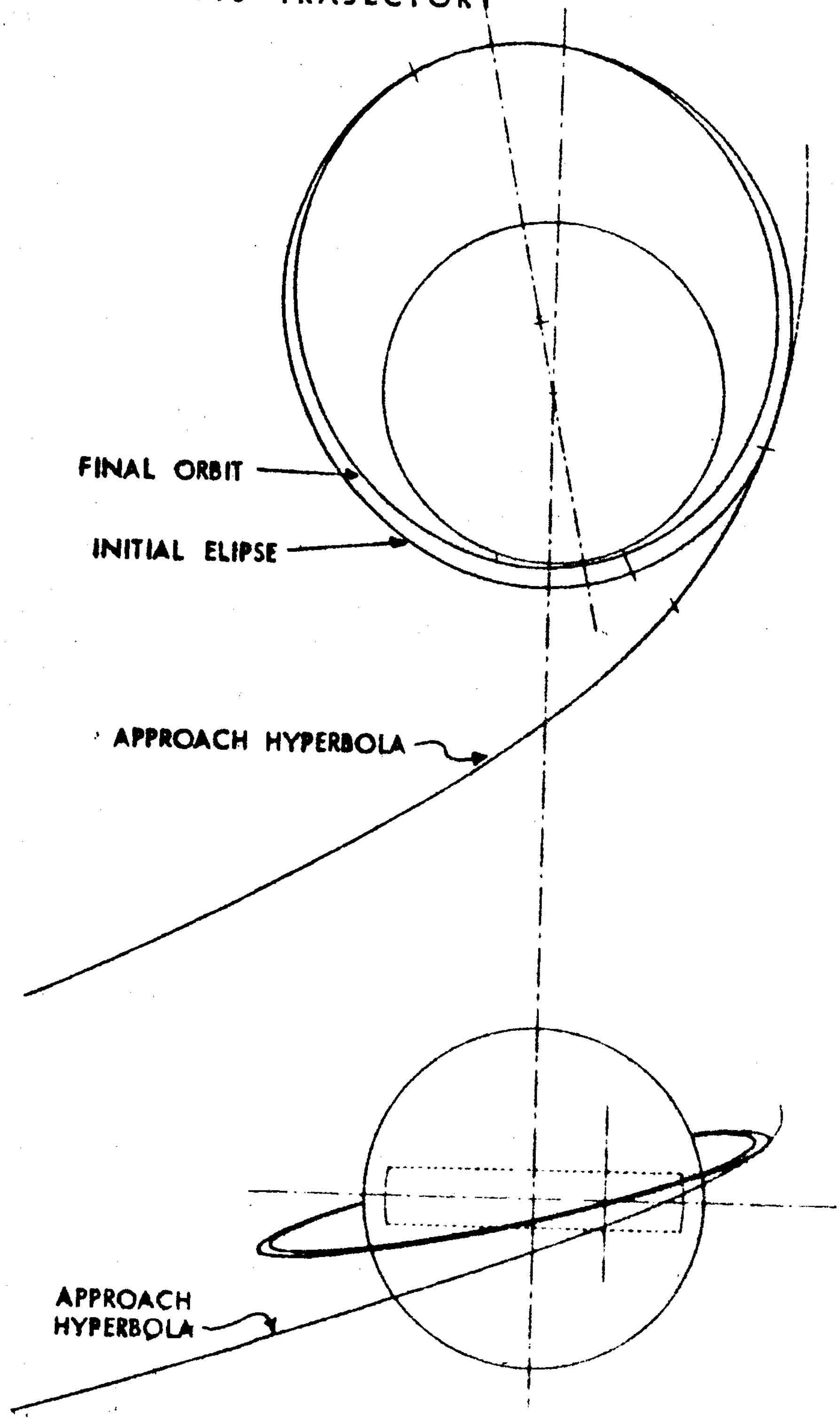

FIGURE 3.1.1.8

TOTAL VELOCITY INCREMENT VERSUS WAITING TIME

JUNE 1966
ASCENDING NODE

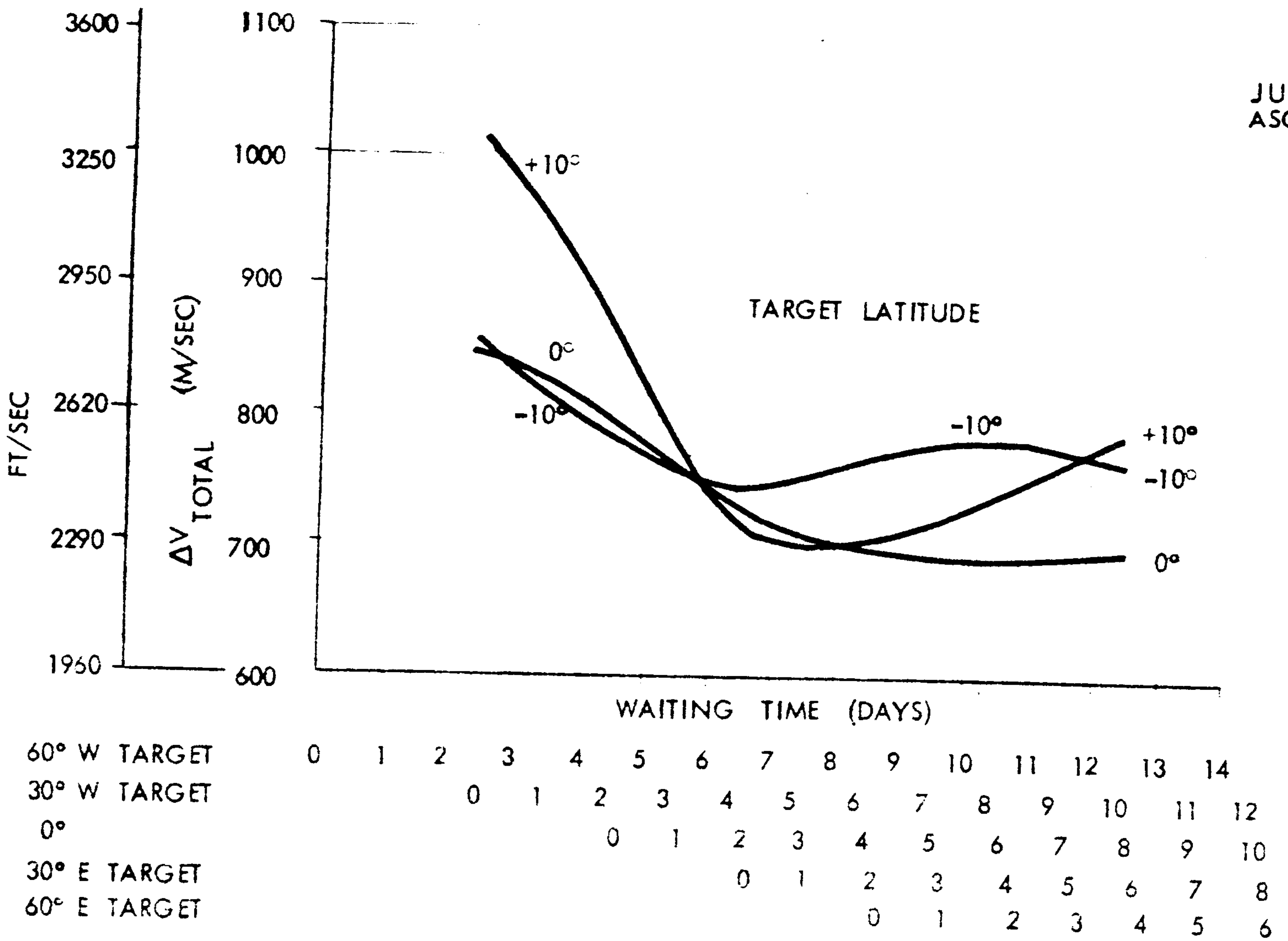

FIGURE 3.1.1.9

TOTAL VELOCITY INCREMENT VERSUS WAITING TIME

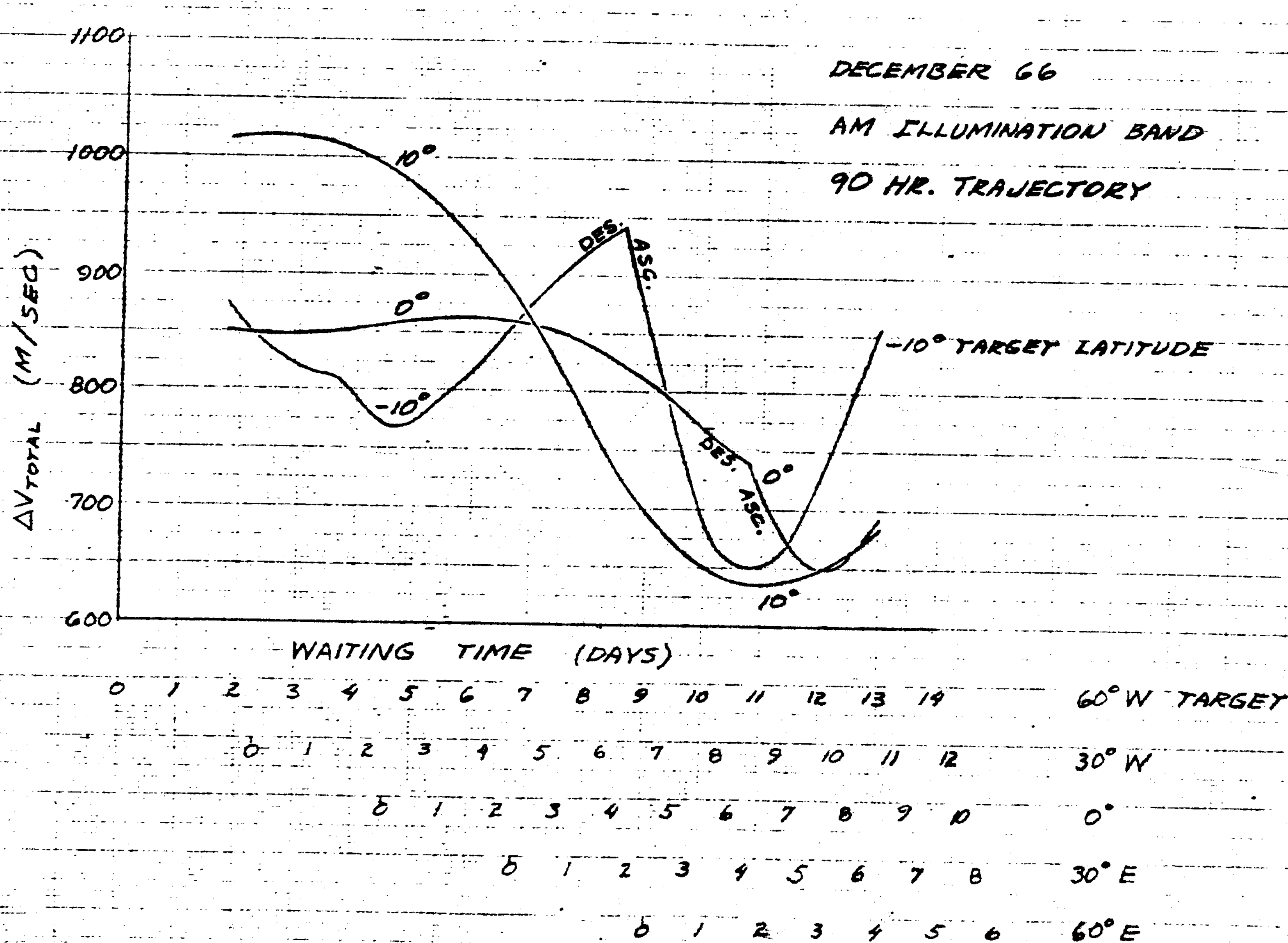

FIGURE 3.1.1.10

3.1.1 (Continued)

to Figure 3.1.1.3 and the penalty associated with the introduction of the solar illumination constraint to the conditions of the experiment. This constraint is not limited to the photography experiment. It applies to any experiment which must be related to sun angle, such as infrared, X-Ray fluorescence or photometry/colorimetry experiments.

It is also to be noted that the effectiveness of controlling the time of arrival at the target area, measured as the time increment gained per degree of plane change, decreases as a function of orbital inclination for a fixed latitude target. A waiting time prior to arrival at a fixed target will therefore, generally exist for high inclination orbits. This time can be usefully exployed to perform experiments in other areas of the lunar surface and does not necessarily represent idle time.

Generally, when an illumination constraint exists, an expenditure of an additional delta-velocity increment of 150-200 meters per second must be provided in order to insure a reasonable launch period (number of days per month when launches are possible) for all targets.

A further penalty may be incurred if it is desired to minimize the probability of impacting the lunar surface because of out of tolerance subsystem performance during midcourse. This would require an initial high altitude aimpoint on initial approach, with a corresponding decrease in injection efficiency, and a requirement

3.1.1 (Continued)

for an orbit transfer velocity increment as shown by the data of Figures 3.1.1.11, 3.1.1.12, 3.1.1.13 and 3.1.1.14 for final orbit perilunes of 50, 100, 150 and 200 km respectively.

3.1.2 Solar Illumination Requirement

Surface oriented experiments, such as, photometry/colorimetry, photography, radiometry, infrared mapping, etc., will be designed to operate within a specified range of solar illumination of the surface. For any single experiment to be performed near the perilune, this implies a constraint on the number of days per month during which the spacecraft can be launched. This constraint becomes more restrictive as the spacecraft capability to control the time of arrival at a specific target area is curtailed. In the limiting case, where no capability for rotation of the line of nodes after arrival at the moon is provided, the launch period is controllable only by variation in transit time which results in a limited line of nodes shift (Figure 3.1.1.1). If a fixed illumination angle requirement and a fixed transit time are assumed, in addition to the assumption of no controllability of the position of the line of nodes after spacecraft arrival at the moon, then the launch period is limited to a single time instant per month. The capability of shifting the line of nodes of the lunar orbit after arrival at the moon (Waiting time control) can be exchanged on a one to one basis for an extension of the launch period (number of possible launch days per month) for these experiments.

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ΔV FOR LUNAR ORBIT TRANSFER

H_{p2} = 50 KM

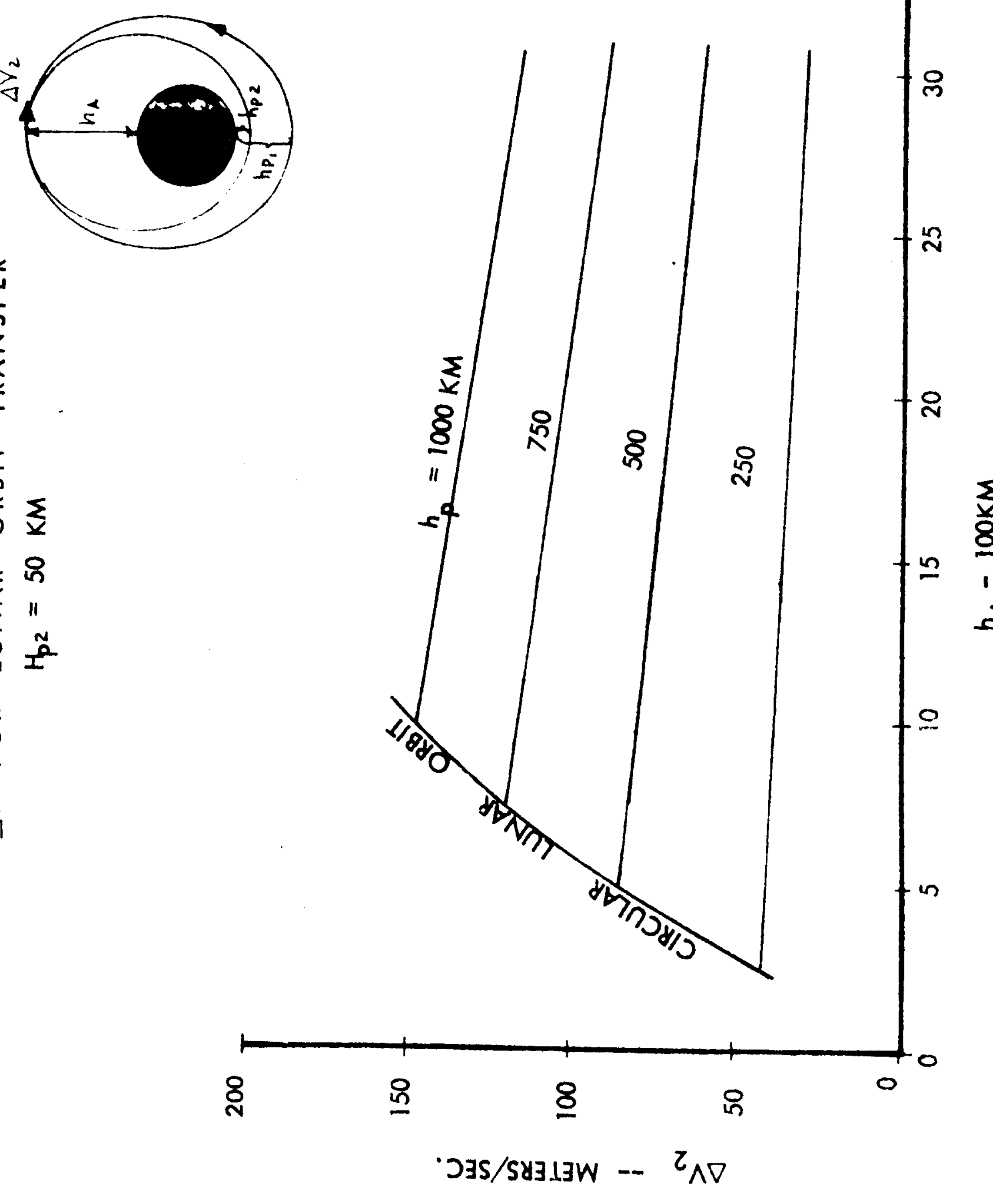

FIGURE 3.1.1.1.11

ΔV FOR LUNAR ORBIT TRANSFER
$H_{P_2} = 100$ Km

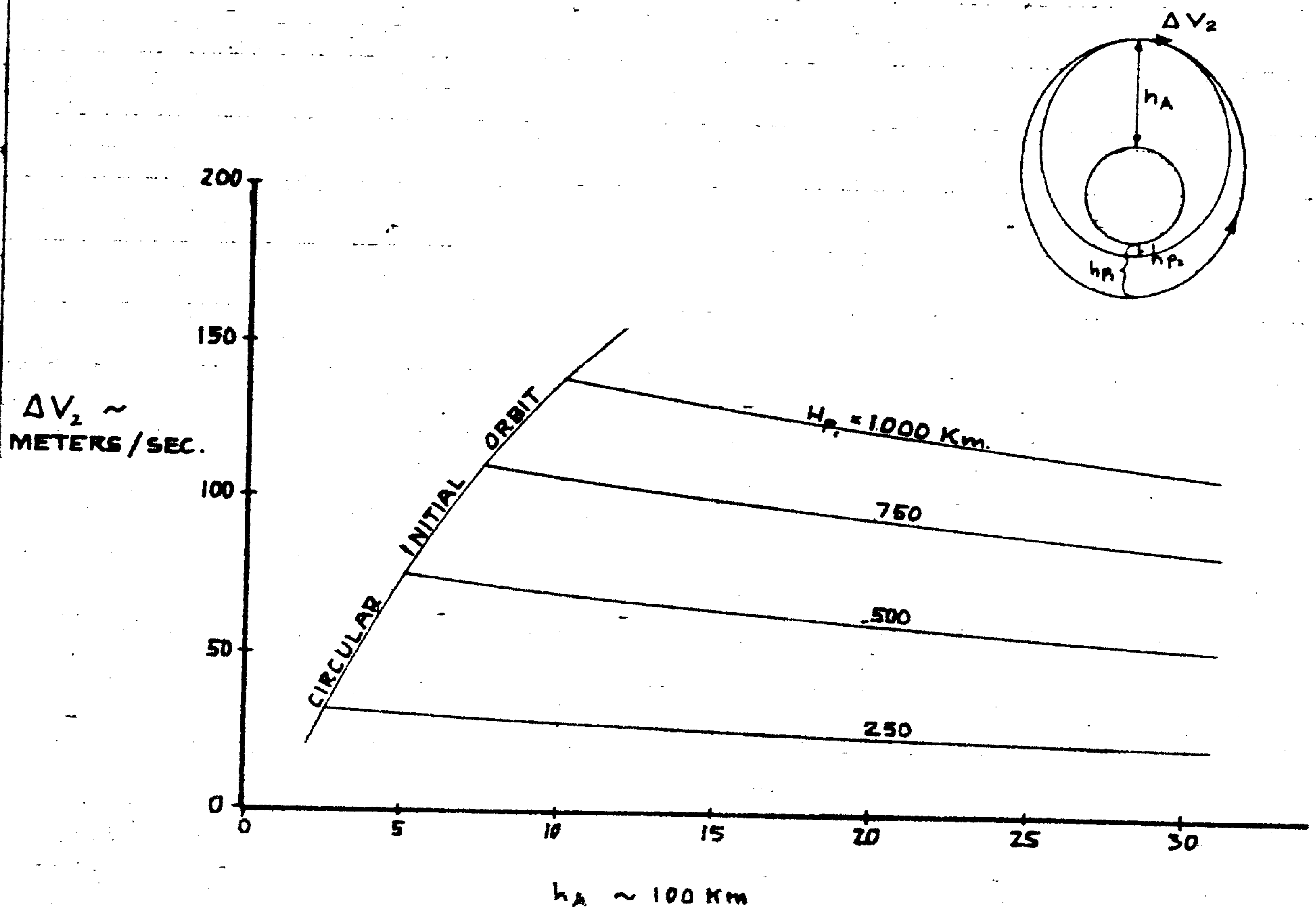

FIGURE 3.1.1.12

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ΔV FOR LUNAR ORBIT TRANSFER

$h_{p_1} = 150$ Km

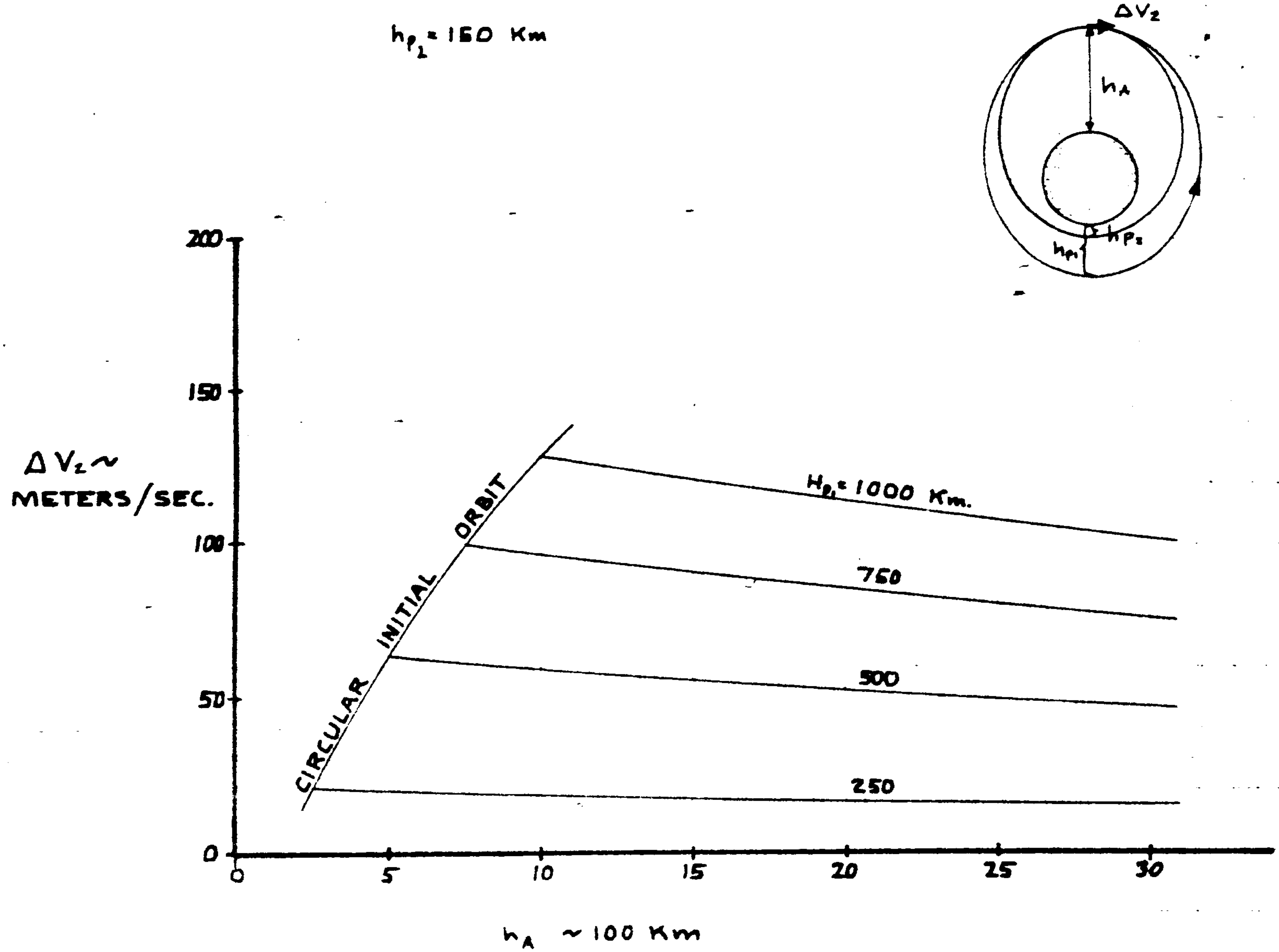

FIGURE 3.1.1.13

ΔV FOR LUNAR ORBIT TRANSFER

H_{P_2} = 200 Km

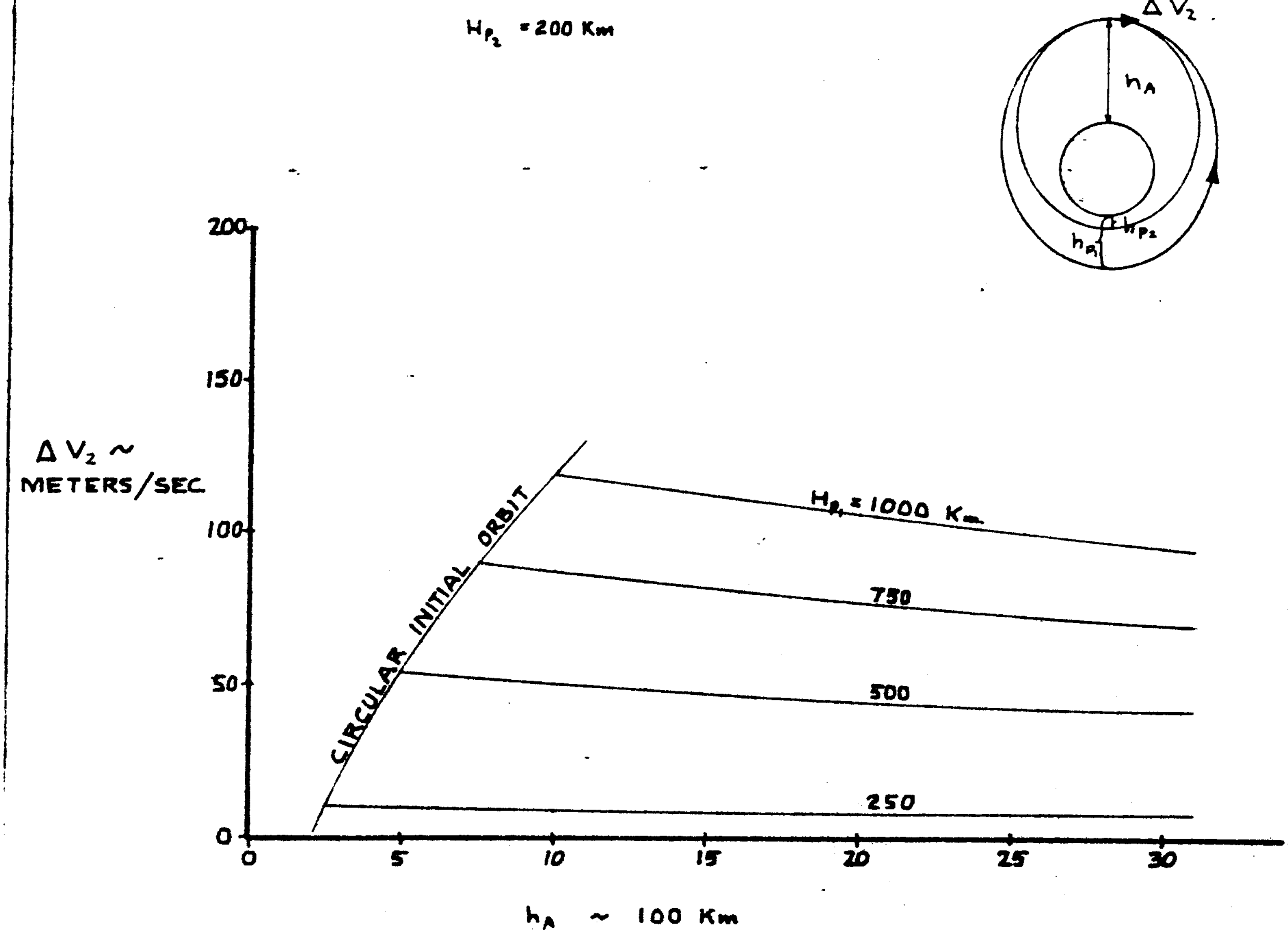

FIGURE 3.1.1.14

3.1.2 (Continued)

Experiments grouped together which have different solar illumination requirements introduce additional mission and experiment conduct constraints since the lunar orbit is to a first approximation inertially fixed. With an inertially fixed orbit, the solar illumination at a given point in orbit remains fixed, and therefore, the experiments requiring different illuminations must be performed at different altitudes depending on the eccentricity of the lunar orbit and the difference in their illumination requirements. Additionally, for orbits with an inclination other than equatorial the latitude coverage of experiments requiring different solar illuminations will generally be different; which would make the correlation of data from a single flight difficult for such experiments. This effect is illustrated in Figure 3.1.2.0 for an arbitrarily inclined orbit with arbitrary solar illumination differences for two experiments. An exception to the above illustration would exist if the mission were specifically designed to achieve coverage of the same latitude by two experiments with different solar illumination requirements. This possible mode of operation is shown in Figure 3.1.2.1. It is to be noted with reference to the figure that if the illumination requirements of the two hypothetical experiments do not differ greatly the orbital inclination would approach target latitude. As a result the width of latitude coverage achievable would decrease to that coverage achievable by a single cross-range scan capability of the experiment. As the differential of illumination increases,

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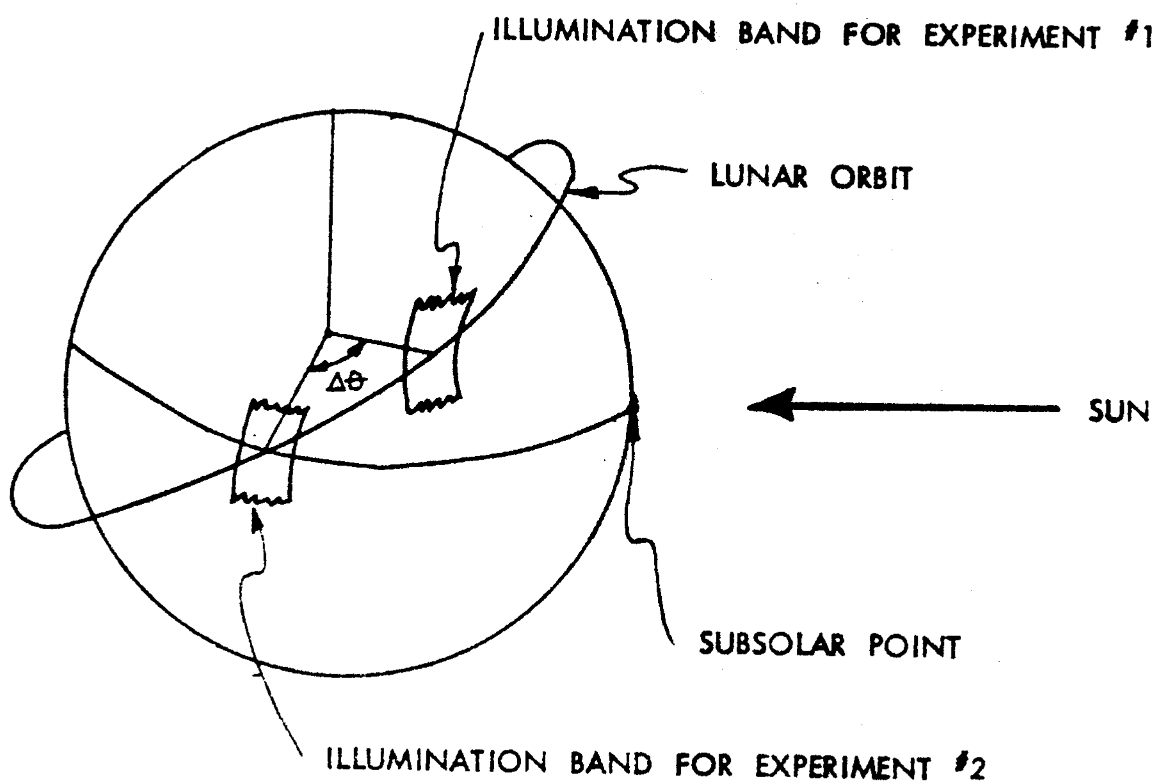
EFFECT OF ILLUMINATION DIFFERENTIAL AND ORBIT INCLINATION



$\Delta\theta$ = CENTRAL ANGLE BETWEEN THE TWO EXPERIMENTS

FIGURE 3.1.2.0

COVERAGE AT SAME LATITUDES
DIFFERENTIAL IN SOLAR ILLUMINATION

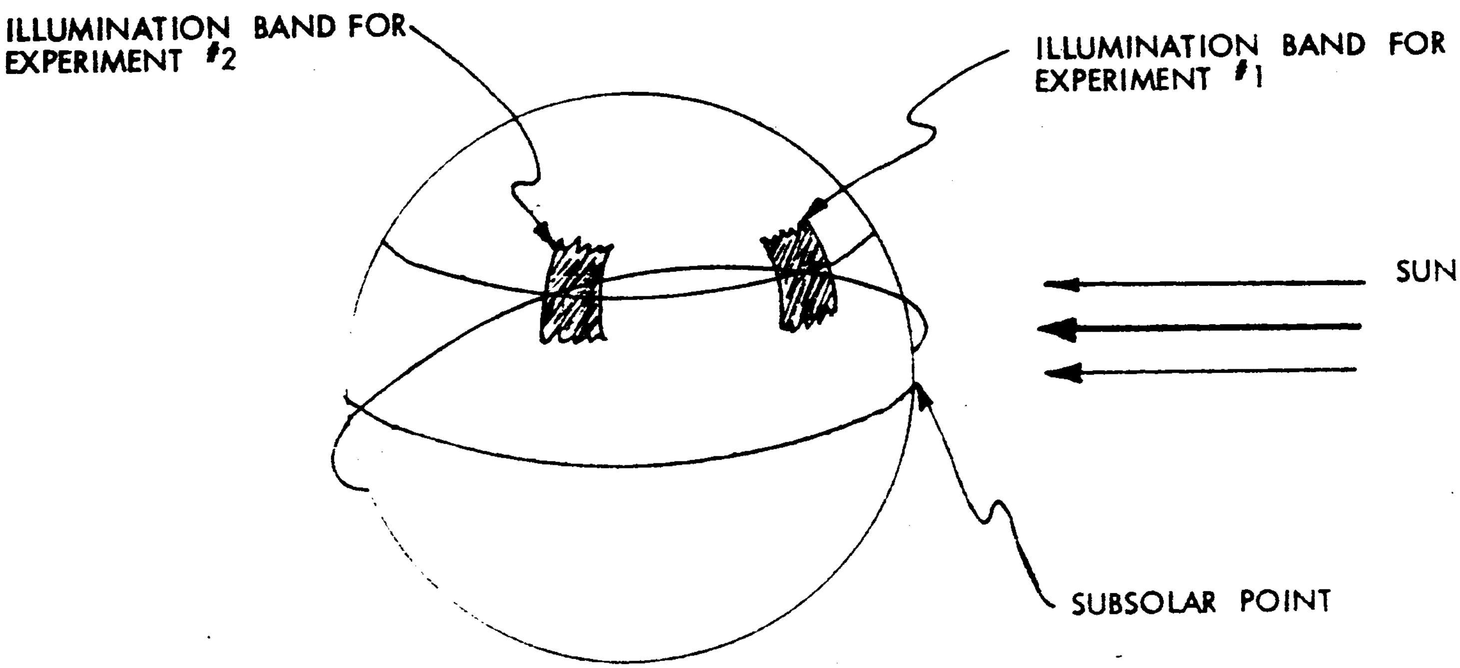


FIGURE 3.1.2.1

3.1.2 (Continued)

a progressively higher differential between the orbit inclination and target latitude is allowable. In the upper limit, at a 180° solar illumination differential, the inclination of the orbit can be as high as 90° in the mode of operation illustrated in Figure 3.1.2.1.

In this case contiguous area coverage can be achieved by the experiment(s), taking advantage of the rotation of the moon with respect to the orbit, provided that the single crossrange scan capability of the experiment(s) is equal to or greater than the distance a point on the lunar surface moved, within a direction normal to the orbit plane, during a single orbital pass. This subject is covered in greater detail in Section 3.1.4.

3.1.3 <u>Altitude of Measurement</u>

The altitude requirements of an experiment, in conjunction with the central angle over which the experiment is to be performed, introduce a constraint on orbit eccentricity. This constraint is expressed by the following relation:

$$a = \frac{R_1 \left(R_1 - R_2 \cos \frac{\epsilon}{2}\right)}{2R_1 - R_2 \left(1 + \cos \frac{\theta}{2}\right)}$$

$$e = \frac{R_2 - R_1}{R_1 - R_2 \cos \frac{\theta}{2}}$$

which fixes both the orbit period and orbit eccentricity for given:

3.1.3 (Continued)

R_1 = Lower limit on experimental altitude (assumed to define perilune, radius R_1)

R_2 = Upper limit on experimental altitude (assumed to define the radius R_2 at a central angle $\frac{\theta}{2}$ from perilune.)

$R_1 = R_{moon} + H_{min.}$

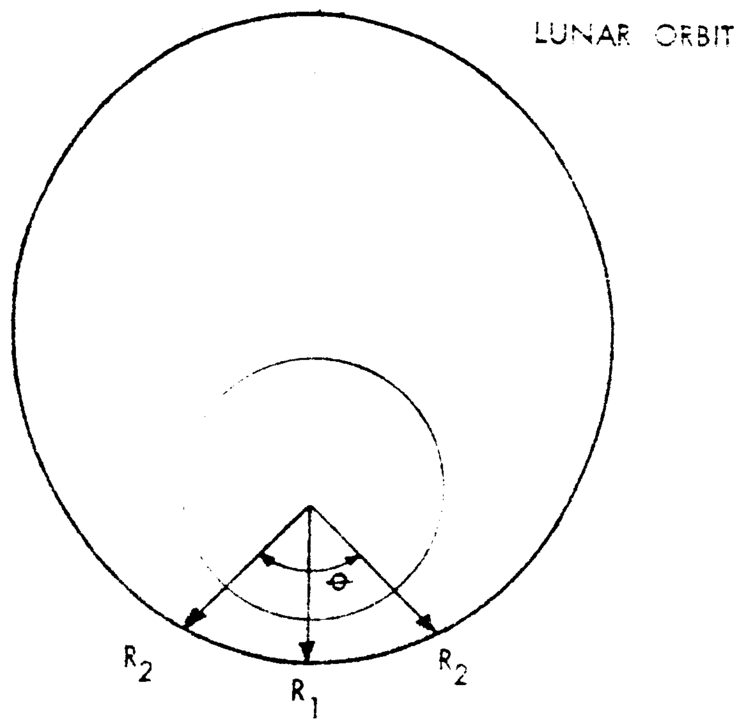
$R_2 = R_{moon} + H_{max}.$

θ = Central angle over which the experiment is to be performed.

The above relation is illustrated in Figure 3.1.3.0.

The preceeding relation can be applied, in a modified form, to orbit design where two separate experiments separated by a given central angle are to be performed. This may be the case, for example, when the central angle spacing between two experiments is due to a required solar illumination angle differential requirement. The relationship shown in the preceeding discussion applies with the modification that the terms R_1 and R_2 should be interpreted as the mean radii at which the two experiments are to be performed and $\frac{\theta}{2}$ represents the central angle spacing between the two experiments. In the above usage of the altitude constraints of the experiments no degree of freedom exists for controlling the range of altitudes over which each of the individual experiments will vary if R_1, R_2 and $\frac{\theta}{2}$ are given.

EXPERIMENT ALTITUDE CONSTRAINTS



$R_1 = R_m + H_{min}$

$R_2 = R_{na} + H_{max}$

H_{min} = MINIMUM EXPERIMENT ALTITUDE

H_{max} = MAXIMUM EXPERIMENT ALTITUDE

Rm = RADIUS OF THE MOON

θ = CENTRAL ANGLE OVER WHICH THE EXPERIMENT IS TO BE PERFORMED

FIGURE 3.1.3.0

3.1.3 (Continued)

The above process can, of course, be carried out in reverse (i.e., with a, e, $\frac{\theta}{2}$ and R_1 given, then R_2 is completely defined) if the experiment of radius R_1 is given priority and other system constraints dictate the orbital parameters. The latter will generally hold if more than two experiments separated in central angle from each other are involved.

3.1.4 Area Coverage and Contiguity

The area coverage capability of an orbiting experiment is dependent on the transverse scan capability of the experiment, length of operation of the experiment during a single orbital pass, orbit period and target location.

If the experiment requires contiguous area coverage then its transverse scan capability must be consistent with the displacement of a given point on the lunar surface during a single, or multiple, lunar orbit. This is illustrated in Figure 3.1.4.0, which establishes the equivalents between the rotation of the surface relative to the orbit and orbit rotation relative to the surface, and shows the relation to the transverse scan requirement.

Data relating crossrange or transverse displacement between successive passes to orbital inclination with target latitude as a parameter are shown in Figures 3.1.4.1 and 3.1.4.2 for

CONTIGUOUS AREA COVERAGE

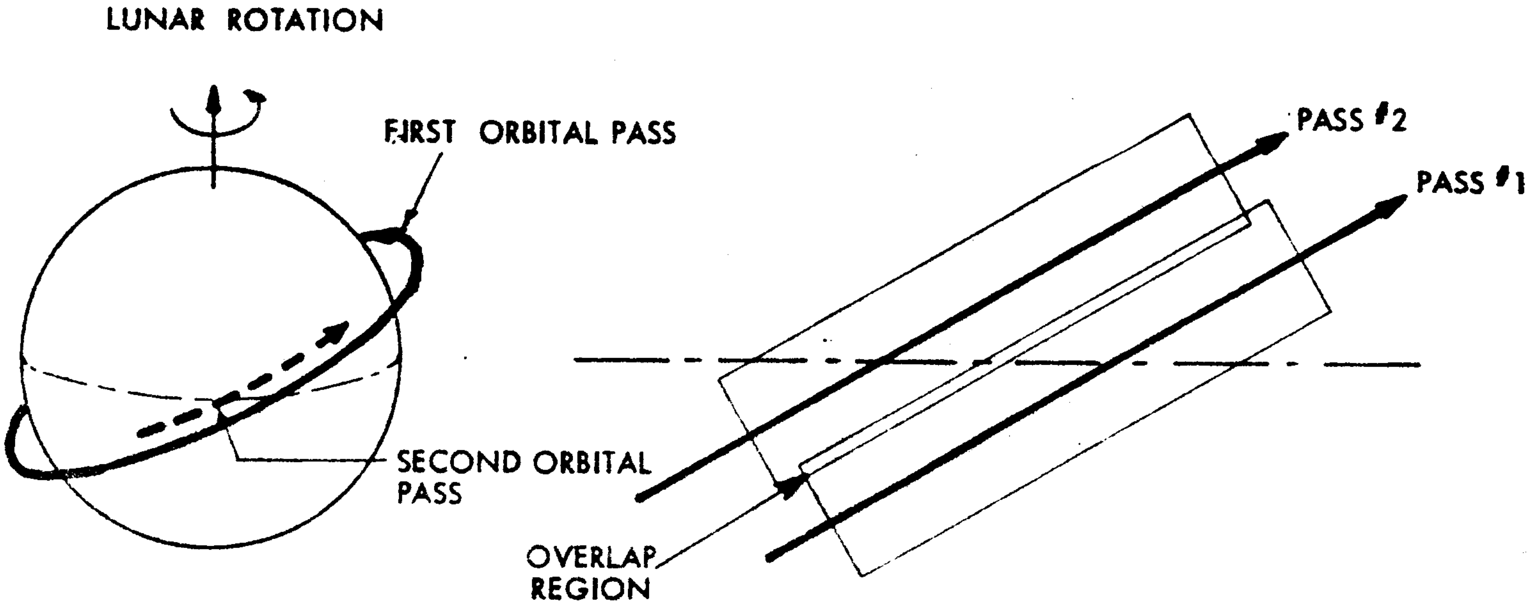


FIGURE 3.1.4.0

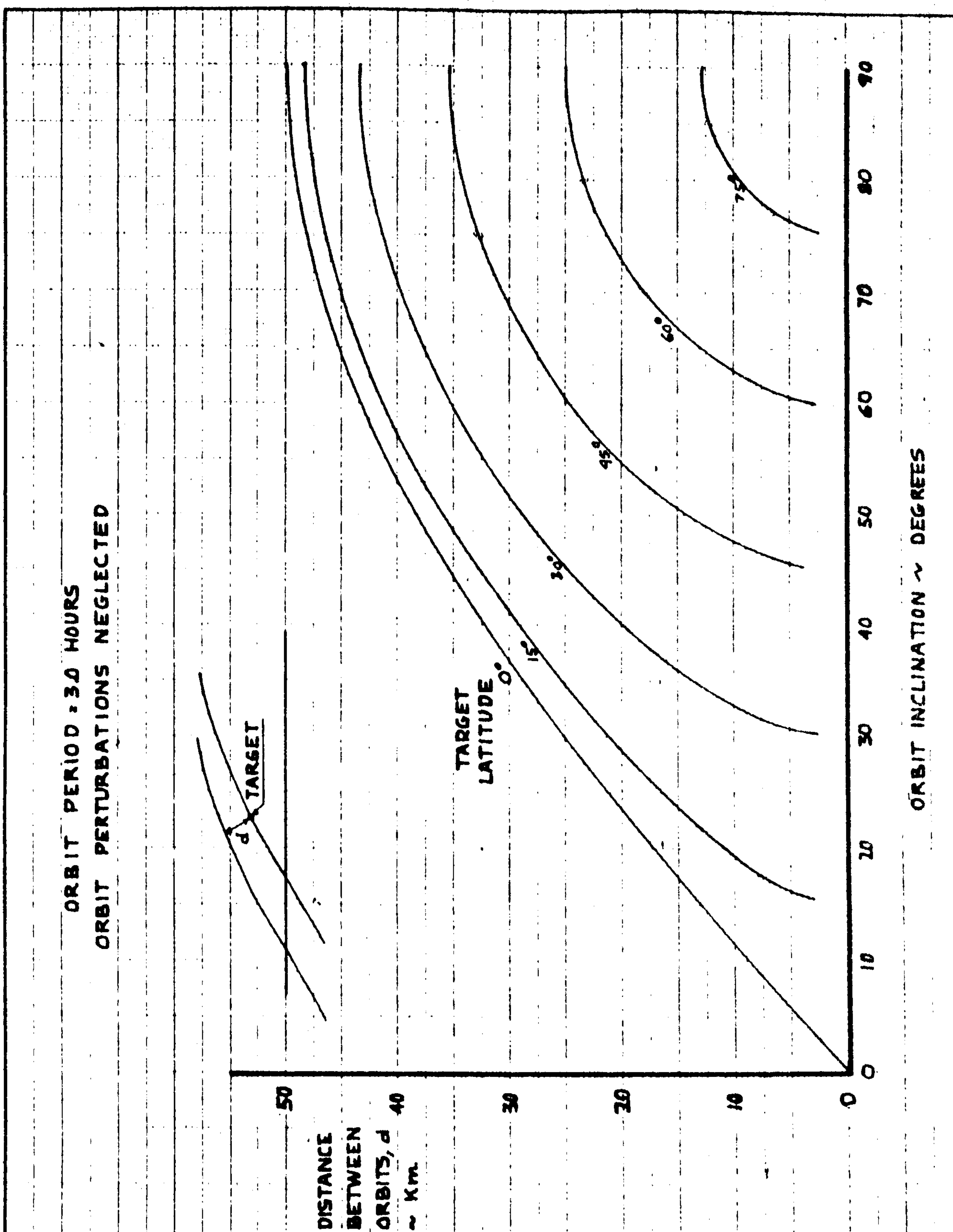


FIGURE 3.1.4.1

	INITIALS	DATE	REV BY INITIALS	DATE	TITLE	MODEL
CALC	R UBAGHS	5-26-66			DISTANCE BETWEEN SUCCESSIVE ORBITAL PASSES ORBIT PERIOD = 3.0 HOURS	
CHECK						
APPD						
APPD						

U3 4013 8000 REV. 12 64

REV LTR

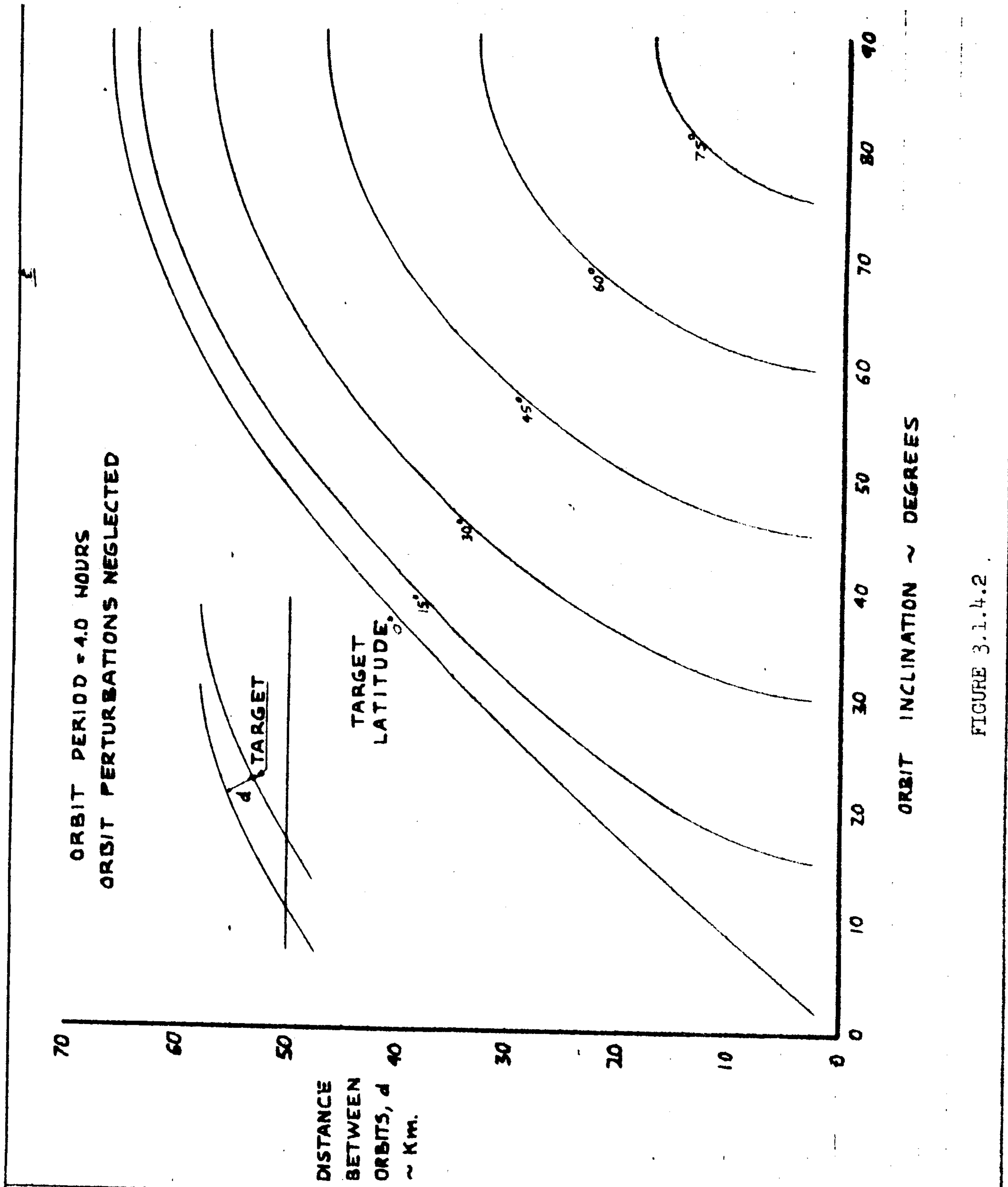


FIGURE 3.1.4.2

	INITIALS	DATE	REV BY INITIALS	DATE	TITLE	MODEL
CALC	R.UBAGHS	5-26-66			DISTANCE BETWEEN SUCCESSIVE ORBITAL PASSES ORBIT PERIOD = 4.0 HOURS	FIG 10
CHECK						
APPD.						
APPD.						

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REV LTR__________

3.1.4 (Continued)

orbital periods of 3 hours and 4 hours respectively. This data can be extrapolated linearly to any other orbital periods and provides basic orbital information for the design of experiment scan systems where coverage contiguity is of interest.

The length of operation of the experimentation of a single orbital pass will be, in addition to constraints resulting from spacecraft subsystem operational requirements to be discussed in subsection 3.2.0, dependent on solar illumination band constraints and altitude constraints discussed in the previous subsections. The altitude constraint is self-evident on the basis of the discussion of subsection 3.1.3. The constraint introduced by a requirement for performing the experiment between given limits of solar illumination is inclination dependent with respect to limitation of the arc length over which the experiment is to be performed. This effect is illustrated in Figure 3.1.4.3.

3.2 SUBSYSTEM TRADE PARAMETERS

This section outlines the general Lunar Orbiter subsystem flexibility trade parameters, growth potential and the constraints generated by these considerations relative to orbital parameters and experiment planning.

3.2.1 Velocity Control Subsystem (VCS)

The nominal Block I Lunar Orbiter velocity control subsystem capability, in terms of available delta velocity increment, as

EXPERIMENT OPERATION ARC LENGTH VS. ILLUMINATION BAND CONSTRAINT AND INCLINATION

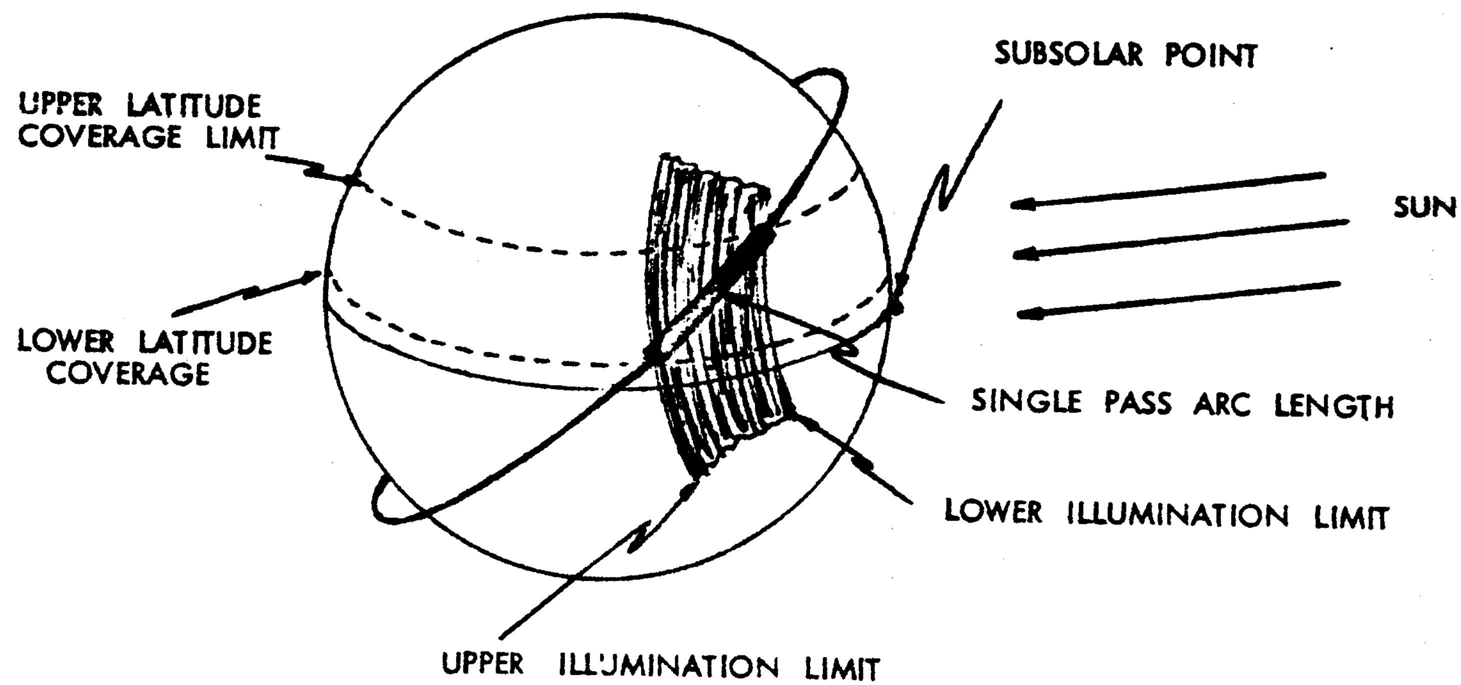

FIGURE 3.1.4.3

3.2.1 (Continued)

a function of spacecraft weight is shown in Figure 3.2.1.0 by reference to the nominal specific impulse curve of 276 seconds.

The above impulsive velocity increment budget does not include the requirements for error corrections and finite burn time. The error correction budget and the velocity increment available for injection into lunar orbit are as shown below:

VELOCITY CONTROL BUDGET

	Item	3σ Value
1.	Midcourse Corrections	75 m/sec.
2.	Injection Correction	50 m/sec.
3.	I_{sp} Decrease to 270 sec.	20 m/sec.
4.	RMS Total	92 m/sec.
5.	Finite Burn	25 m/sec.
6.	Total Preinjection Budget	117 m/sec.
7.	Lunar Orbit Injection and Transfer	863 m/sec.
8.	Total available	980 m/sec.

The allocation of 863 m/sec. for lunar orbit injection constitutes a sufficient provision for injection into a circular low altitude lunar orbit as can be seen by reference to Figure 3.1.1.3. This capability is based on the simplifying assumption that no experiment constraints; such as solar illumination, target position, etc., need to be considered and that operational launch constraints don't introduce additional velocity requirements. A

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ΔV & LAUNCH VEHICLE CAPABILITY VS SPACECRAFT WEIGHT

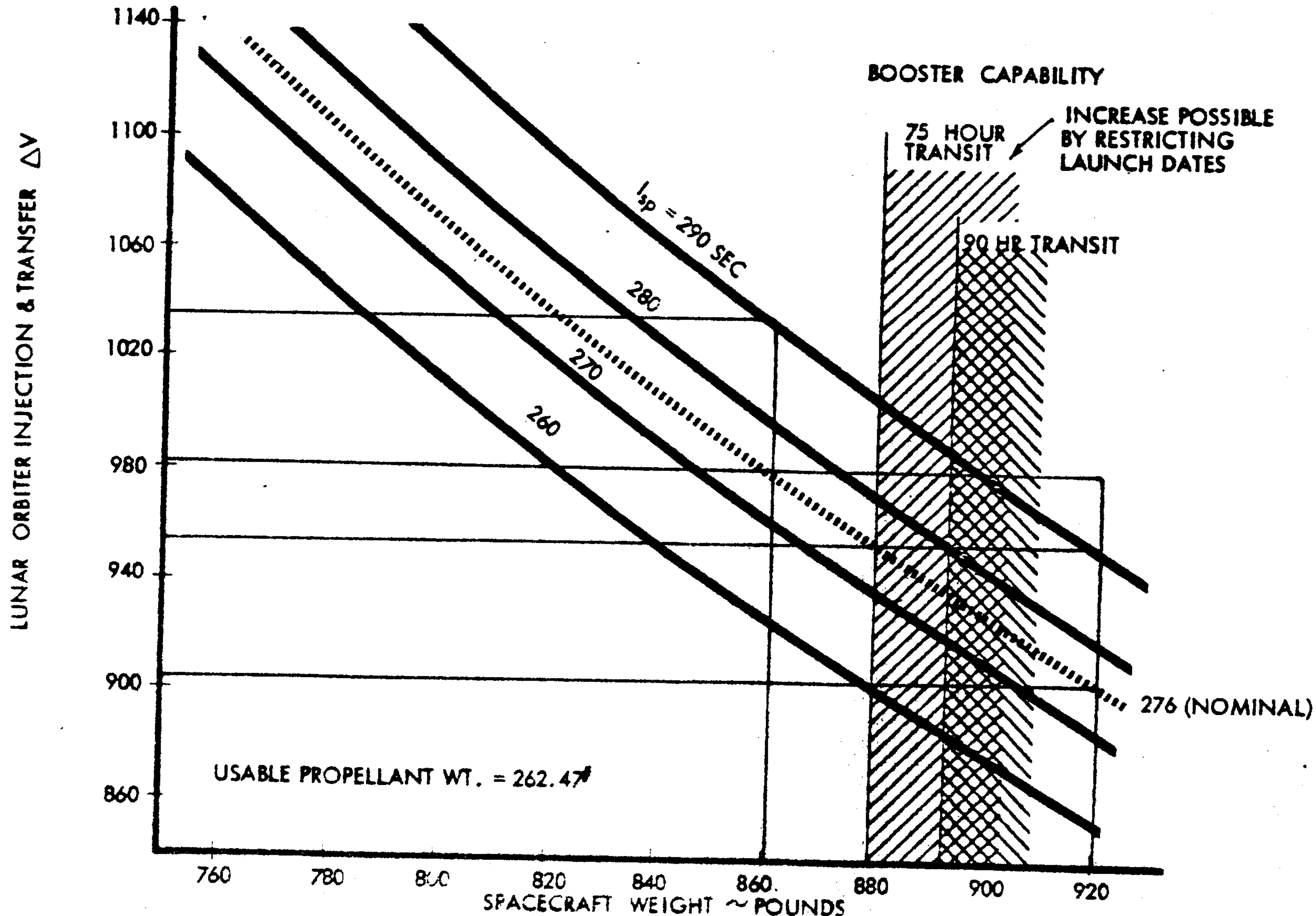

FIGURE 3.2.1.0

3.2.1 (Continued)

more realistic assessment, including the above factors, will be shown subsequently relative to consideration of elliptical orbits.

If the spacecraft weight is increased to 920 lbs., as indicated in Case 1 of the L-5382 Statement of Work, then the above budgeting does not allow for injection into circular orbits at altitudes of less than approximately 700 km altitudes as shown in Figure 3.1.1.3. In this case the velocity subsystem, assuming no modification, introduces a constraint of using elliptical orbits if a low experiment altitude is desired.

An increase in velocity control subsystem performance, if desired, is available by the modifications described in succeeding paragraphs.

Spacecraft on-board propulsive capability is a function of the rocket engine's specific impulse and the amount of useable propellant available. The velocity increment capability of a spacecraft is defined by:

$$V = g\, I_{sp} \ln \frac{W_i + W_p}{W_i}$$

where

I_{sp} = specific impulse, seconds

W_i = spacecraft inert weight (including unuseable propellant), lbs.

W_p = useable propellant, lbs.

3.1.2 (Continued)

Thus the spacecraft performance may be improved by increasing specific impulse, increasing the amount of useable propellant, and/or reducing the spacecraft inert weight.

The following paragraphs discuss the specific impulse improvement potentials, and also propulsion system improvements in terms of useable propellant and spacecraft inert weight.

The Marquardt MA-109 rocket engine has been designed and developed for the Apollo program; specifically, for the Service Module and LEM attitude control systems. The MA-109 engine configuration was tailored to the Apollo program requirements so that it has a lower delivered specific impulse than other contemporary engines. The lower performance results from the fact that a pre-igniter chamber is included to minimize over-pressure transients at ignition. The presence of this "foreign body" in the combustion chamber, and its film coolant requirements, account for the reduced performance. The ignition transient may also be eliminated by sequencing the propellant valves such that there is a fuel lead into the engine. This approach is inefficient for engine operated predominently in short pulses such as it is in the Apollo mission; hence, the pre-igniter chamber was incorporated.

The Lunar Orbiter mission does not require pulse mode operation of the engine, and it is entirely feasible to utilize the "fuel lead" configuration and realize a significant increase in specific impulse. This modification would require a qualification test program.

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3.2.1 (Continued)

Engine performance may also be increased (but by a lesser amount) by operating at a different mixture ratio, and/or by increasing the nozzle expansion ratio. The table below summarizes potential engine performance improvements resulting from configuration and operating point changes.

POTENTIAL ENGINE PERFORMANCE IMPROVEMENT

	Configuration	Specific Impulse, Sec. Nominal	Specific Impulse, Sec. Minimum
1.	Present Design	276	270
2.	Shift Mixture Ratio to 1.95	278	272
3.	Increase Expansion Ratio to 60:1	280	274
4.	Combine Items 2 and 3	282	276
5.	Fuel-Lead Configuration (alone)	294	288
6.	Combine Items 5 and 2	296	290
7.	Combine Items 5 and 3	298	292
8.	Combine Items 5, 2 and 3	300	294

It must be emphasized that any of the above changes would require an engine requalification program.

Spacecraft performance may also be improved by decreasing the inert weight and/or increasing the quantity of useable propellant. The present VCS design point is such that the capacity of the fuel tankage is not utilize to maximum capacity. The utilization of maximum propellant tank capacity is equivalent to operating at a mixture ratio of 1.95. Figure 3.2.1.1 shows the effect of engine mixture ratio on spacecraft velocity increment capability.

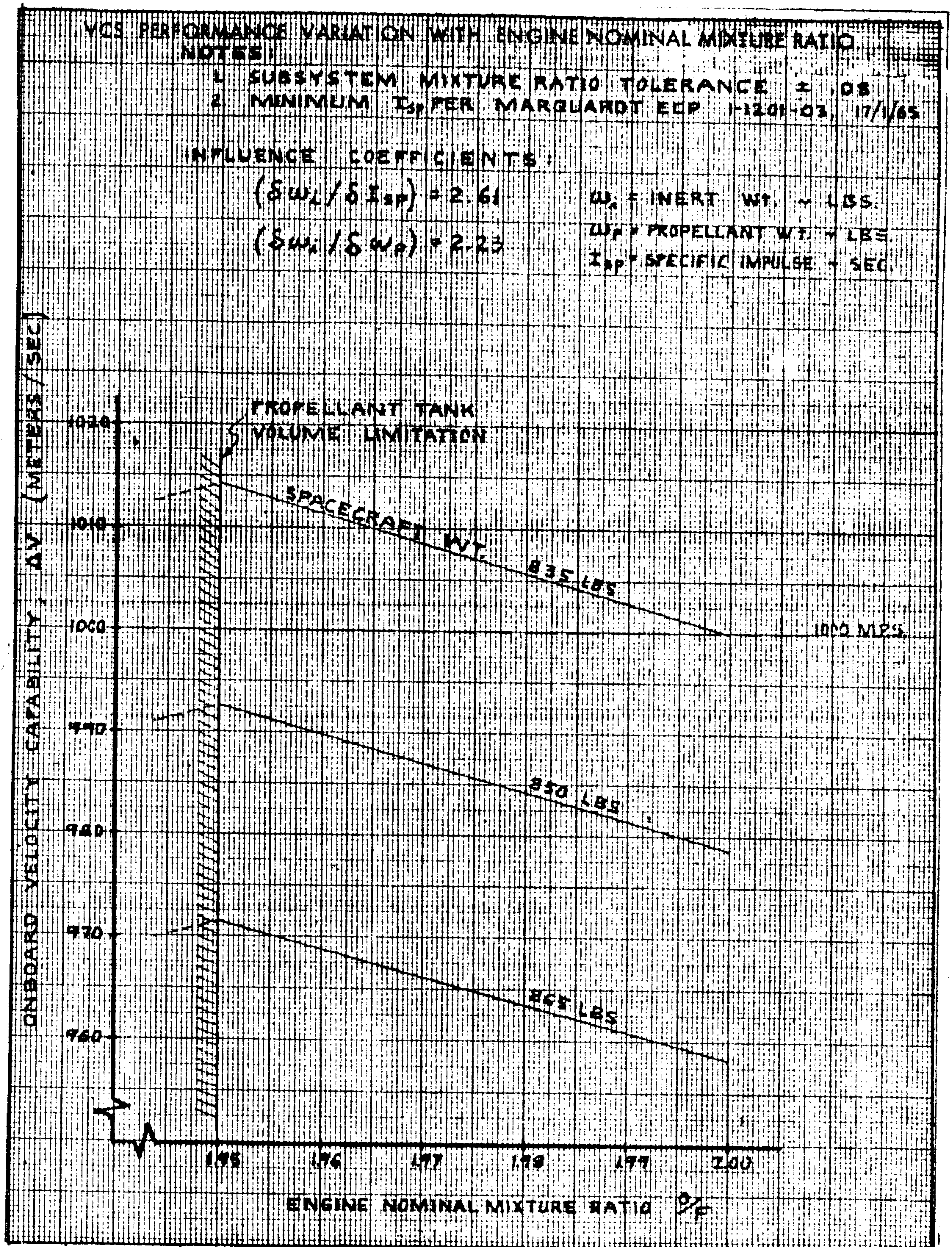
VCS PERFORMANCE VARIATION WITH ENGINE NOMINAL MIXTURE RATIO
NOTES:
1. SUBSYSTEM MIXTURE RATIO TOLERANCE ± .05
2. MINIMUM I_{sp} PER MARQUARDT ECP 1-1201-03, 17/1/65
INFLUENCE COEFFICIENTS:
$(\delta w_i / \delta I_{sp}) = 2.61$
$(\delta w_i / \delta w_p) = 2.23$
w_i = INERT WT ~ LBS
w_p = PROPELLANT WT ~ LBS
I_{sp} = SPECIFIC IMPULSE ~ SEC
PROPELLANT TANK VOLUME LIMITATION
SPACECRAFT WT.
835 LBS
850 LBS
865 LBS
1000 MPS
ONBOARD VELOCITY CAPABILITY, ΔV (METERS/SEC)
1020
1010
1000
990
980
970
960
1.95
1.96
1.97
1.98
1.99
2.00
ENGINE NOMINAL MIXTURE RATIO O/F

FIGURE 3.2.1.1

3.2.1 (Continued)

Observe that for an initial spacecraft weight of 850 lbs., shifting the mixture ratio from 2.00 to 1.95 increases the velocity capability by [illegible] [illegible]/second, a little better than 1%. At even lower mixture ratios, engine performance will continue to increase, but the oxidizer tanks are no longer being utilized to maximum capacity. The resulting degradation in spacecraft performance is indicated by the dashed lines.

Figure 3.2.1.1 also presents two influence coefficinets: these values indicate that, in terms of resultant spacecraft performance, a unit reduction of spacecraft inert weight is over twice as effective as a unit increase in either specific impulse or propellant.

To achieve a significant increase in the quantity of useable propellant requires the installation of larger tankage. By combining various Apollo-program positive expulsion tankage the quantity of useable propellant may be increased to as much as 600 lbs. These tank data, and performance capability in terms of specific impulse, useable propellant, and spacecraft initial weight, are presented in Figure 3.2.1.2.

The potential spacecraft performance, resulting from a retrofit program for the fuel and oxidizer tanks represents a growth potential contingent on increased performance of the translunar boost vehicle as can be seen by reference to Figure 3.2.1.0.

PROPELLANT CAPACITY GROWTH

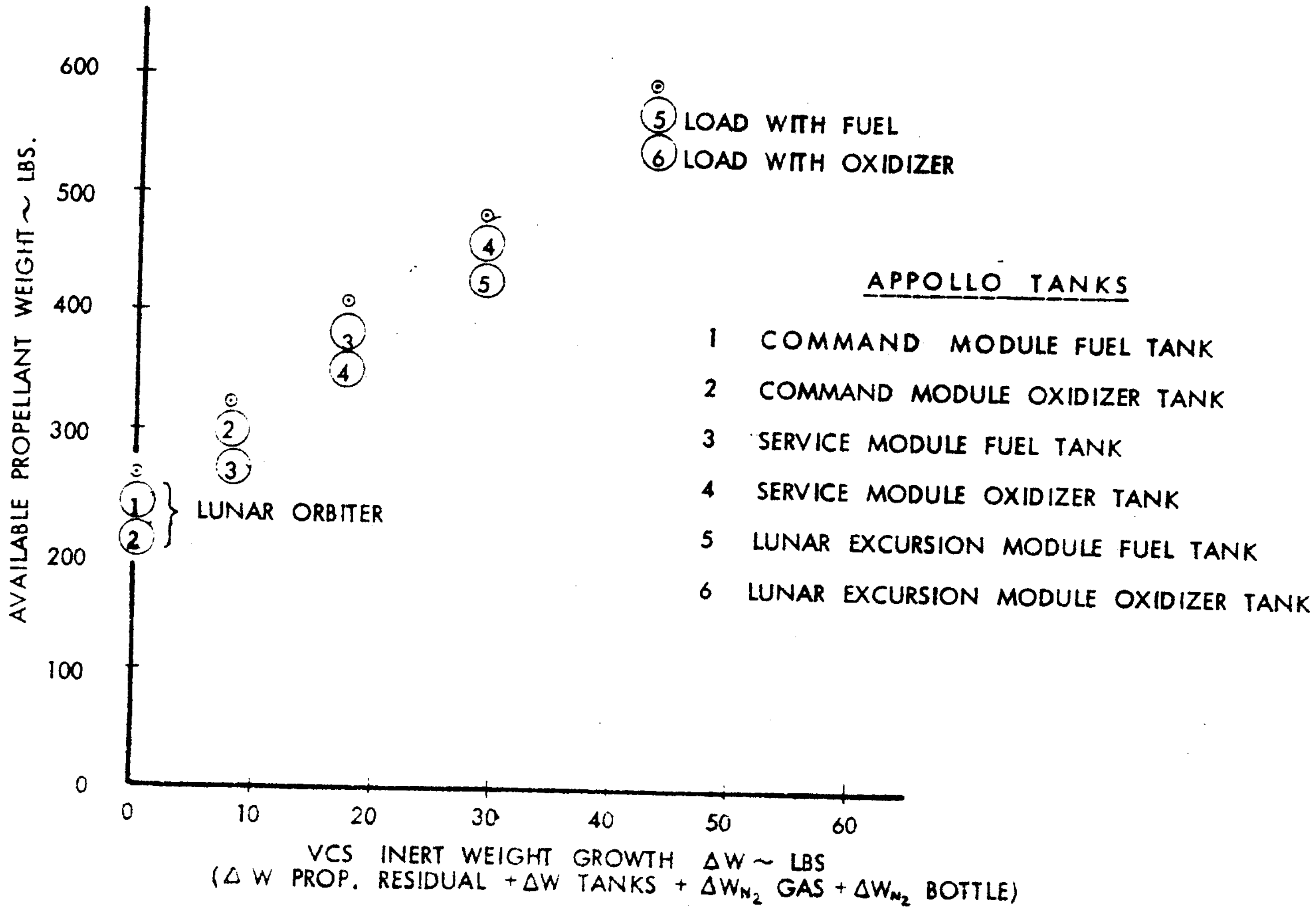

FIGURE 3.2.1.2

3.2.1 (Continued)

The increase in boost capability potential would have to be significant enough to absorb the increase in inert weight and propellant and at the same time provide additional performance.

A case in point is the potential increase of translunar weight capability to 920 lbs. as specified by the L-5382 Statement of Work under Case 1, (Present photographic capability retained). In this case the next increment in flight qualified tanks would result in the addition of 7.5 lbs. of inert weight. An addition of approximately 15 lbs. of propellant (by reference to Figures 3.2.1.3, 3.2.1.4, and 3.2.1.5) would raise the performance level of the velocity control subsystem to a point equivalent to the performance achievable with a spacecraft weight of 860 lbs. without tankage retrofitting. The resulting experimental payload would be decreased from the potential value of 60 lbs. (920-860) to 37.5 lbs. with an additional decrease, not accounted for here, due to increased structure weight.

On the basis of the above examples and the trends shown in Figures 3.2.1.3 through 3.2.1.5 it appears that tankage retrofitting would be justified only if major performance improvements in terms of velocity increment and/or experiment weight carrying capability were desired. For example, a translunar payload capability of 1350 lbs. (approximately the capability of the Atlas/Agena SLV-3X) in conjunction with the propellant capacity increase to 600 lbs. and specific impulse of 290 seconds could

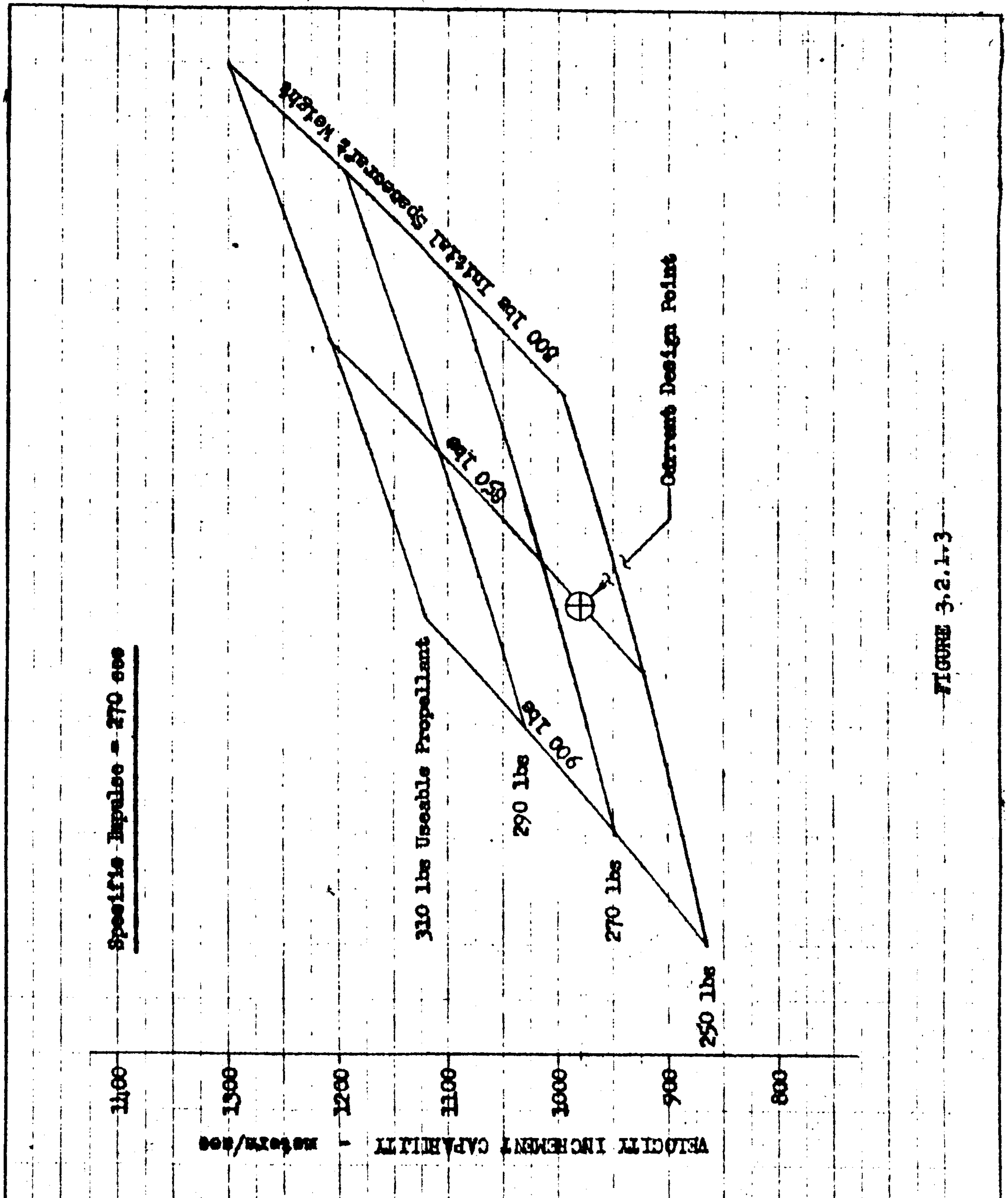

FIGURE 3.2.1-3

Figure 1

	INITIALS	DATE	REV BY INITIALS	DATE	TITLE	MODEL
CALC	JCO	5-7			SPACECRAFT PERFORMANCE	
CHECK						
APPD						
APPD						

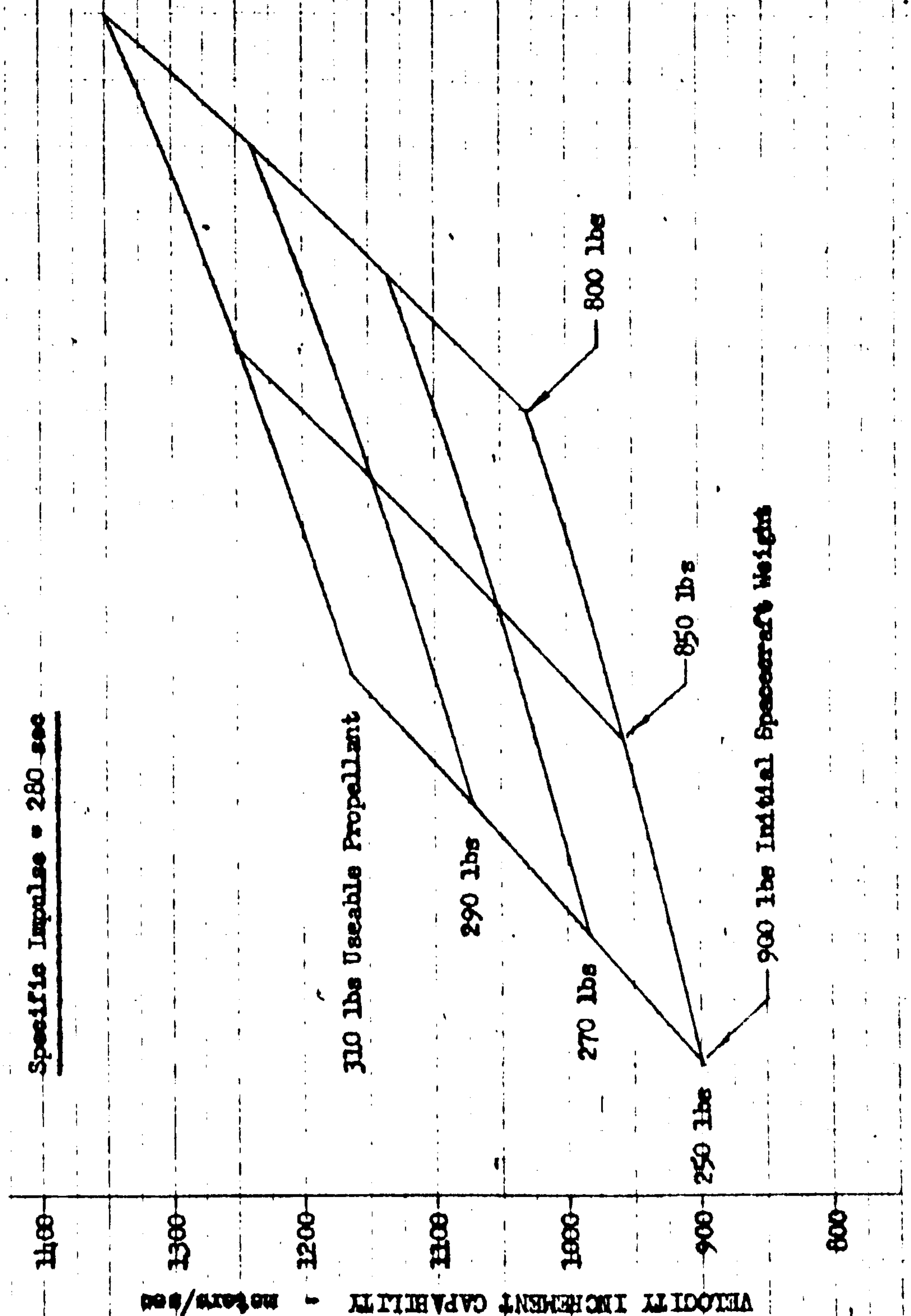

FIGURE 3.2.1.4

Figure 2

	INITIALS	DATE	REV BY INITIALS	DATE	TITLE	MODEL
CALC					SPACECRAFT PERFORMANCE	
CHECK						
APPD						
APPD						

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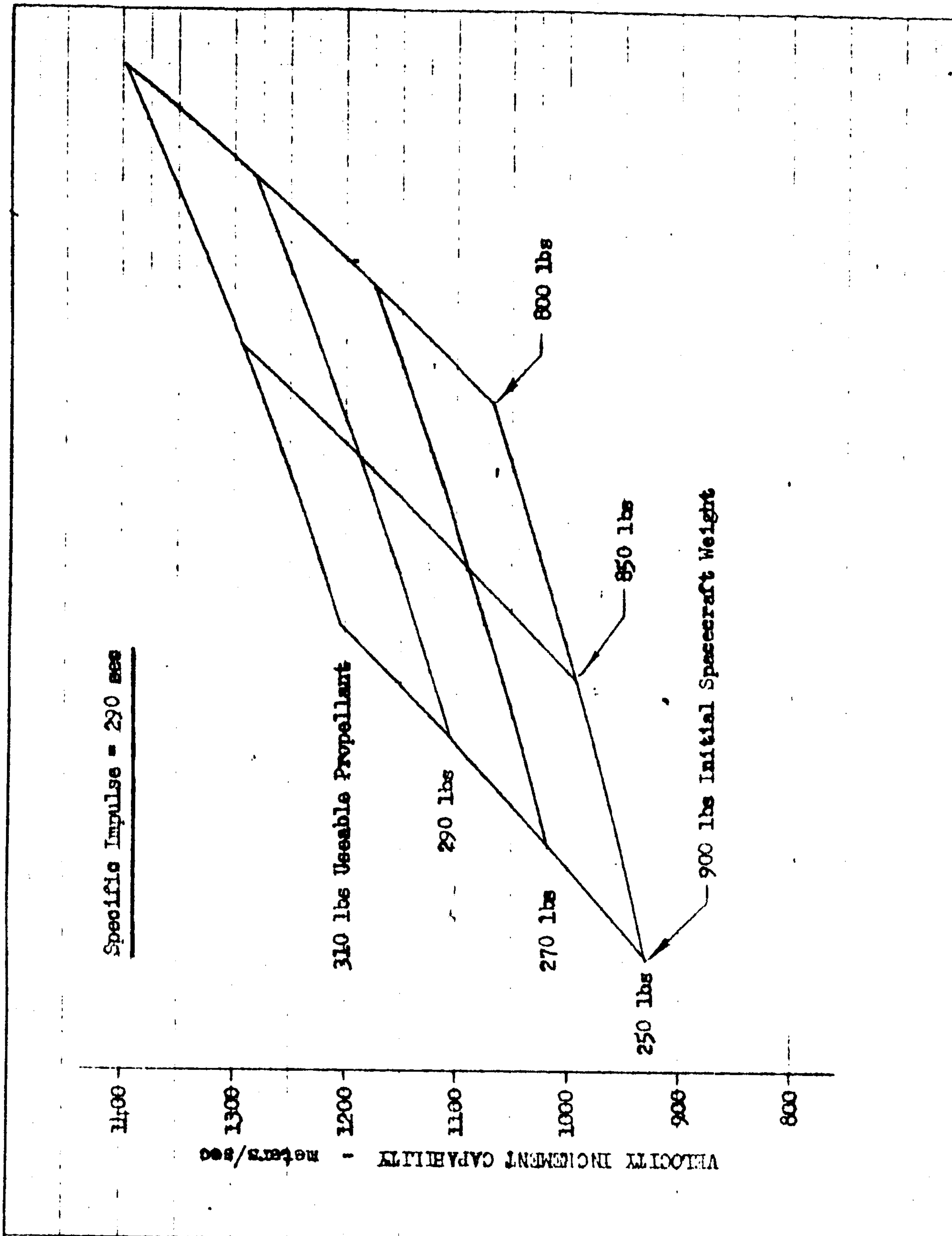

FIGURE 3.2.1.5

	INITIALS	DATE	REV BY INITIALS	DATE	TITLE
CALC	JCC	5-7			SPACECRAFT PERFORMANCE
CHECK					
APPD					
APPD					

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3.2.1 (Continued)

result in an increase in available velocity increment to 1680 m/sec. and an additional payload capability of 90 lbs. for a total of 250 lbs. payload. Alternately this capability could be converted to carrying a total payload of experiments of approximately 400 lbs. with a propellant capacity of 400 lbs. and the same velocity subsystem performance capability as the present system.

Engine perfomance improvement to a specific impulse of 290 seconds minimum would provide velocity increment performance capability for the 920 pound spacecraft equivalent to that of the current minimum specific impulse and an 860 lb. spacecraft. Similarly, the specific impulse improvement is convertible to roughly 15 lbs. additional payload for the 860 lb. spacecraft by fuel off loading.

In addition to the growth potential of the velocity control subsystem, described in the preceeding paragraphs, a significant capability for experiment payload exists if intermittent experiment operation is acceptable. As mentioned in subsection 3.1 intermittent experiment operation may be a requirement imposed by the experiment itself (as is the case of illumination constraints). Additionally, other subsystem constraints, to be discussed in subsequent paragraphs, make intermittent operation of experiments highly desirable.

3.2.1 (Continued)

If intermittent experiment operation is accepted as the mode of operation then a circular orbit does not offer any specific advantages over an elliptical orbit unless multiple experiments need to be performed from nearly the same altitude at different points of a lunar orbit. The latter case can be handled by assignment of experiment priority, with lower priority experiments performed at a non-optimal altitude, using elliptical orbits as discussed in subsection 3.1.3.

The advantage accrued from the acceptance of elliptical orbits is evident, to a first approximation, from Figure 3.1.1.3. The data shown in Figure 3.1.1.3 indicates, for example, that a differential of 295 m/sec. in velocity requirements exists between a circular orbit at 50 km and an elliptical orbit with an apolune of 3000 km, and a perilune of 50 km, at which a given experiment could be performed. This velocity increment differential is convertible into a potential propellant off loading of 72 lbs. The propellant weight decrease is, in turn converti-ble on a one to one basis to an additional experimental payload capacity of 72 lbs.

The above superficially available capacity has to be decreased because a budget of up to 200 m/sec. has to be allowed for the control of position of the perilune of the elliptical orbit, orbit adjustment and provision of an adequate launch period if constraints such as solar illumination of the surface at

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3.2.1 (Continued)

perilune are included. The above consideration reduces the propellant off-loading capability, convertible to experimental payload capability, to 21 lbs. for the 860 pound spacecraft.

The capability for converting propellant weight into useful experimental payloads will range between the limits of 72 lbs. to 21 lbs., for the above change from the circular orbit concept to elliptical orbit concept and will largely depend on experiment operational constraints. Definitive data relating to the conversion of propellant capacity into experimental payload, can be generated only when experimental constraints and corresponding mission parameters are defined. This type of data is shown in Section 4.0. The approximate operational limits of the velocity control subsystem with respect to orbit parameters are shown in Figure 3.2.1.6 for the cases of the previously discussed nominal fuel budget and an adjusted budget including an allowance for operational constraints of 200 m/sec. respectively. A choice of feasible orbits, using the ground rules of the figure, can be translated into potential payload capability by either interpolation in Figures 3.2.1.3, 3.2.1.4 and 3.2.1.5, or the relation

$$W = 264 - W_o \left(1 - e^{\frac{\Delta V}{g I_{sp}}}\right)$$

where

W = Potential payload increment

W_o = Initial spacecraft weight

ΔV FOR LUNAR ORBIT INJECTION AT PERILUNE AND TRANSFER TO 50 km
VS VCS PERFORMANCE

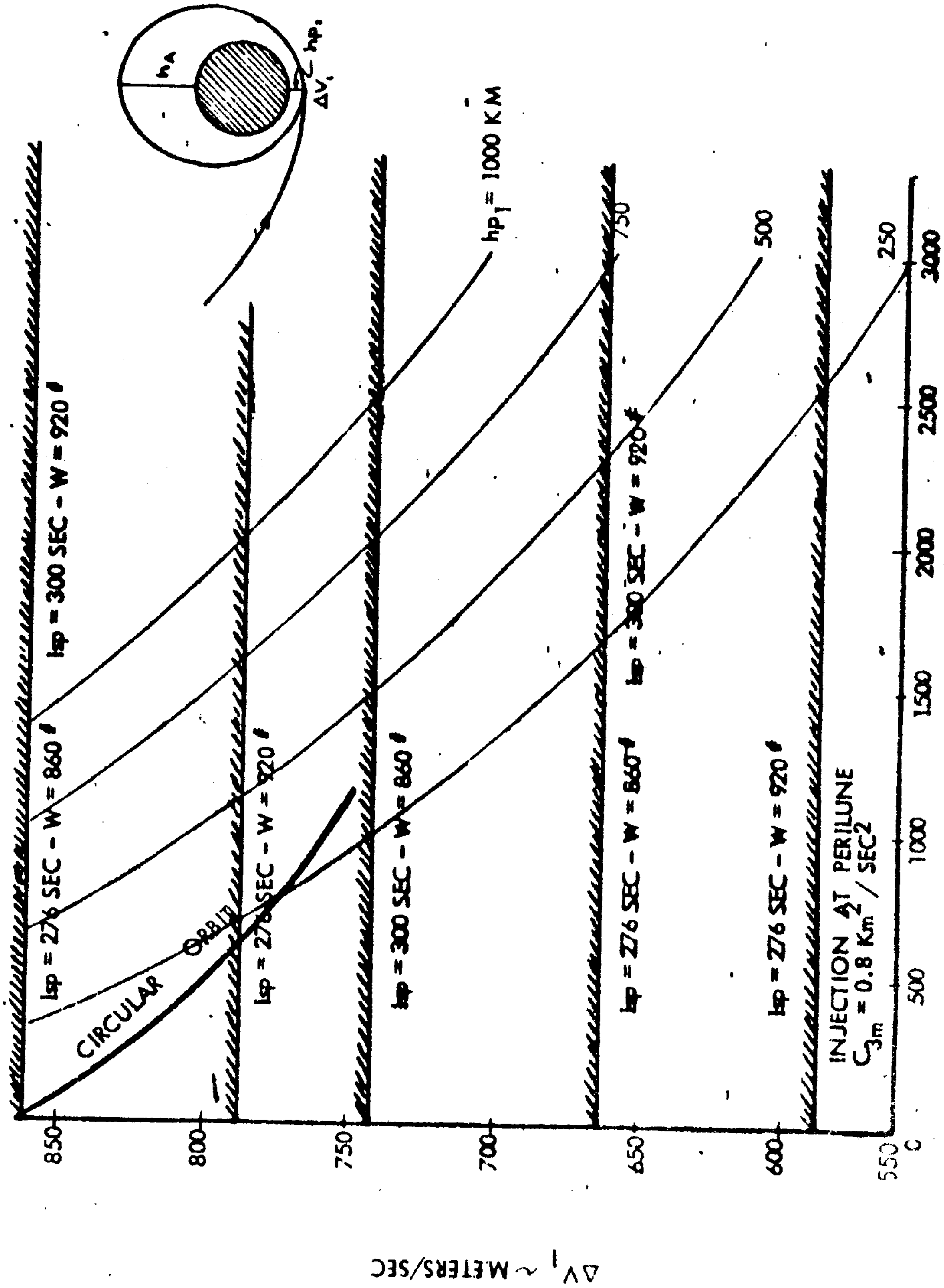

FIGURE 3.2.1.6

3.2.1 (Continued)

ΔV = Velocity increment required (Figure 3.2.1.6)

$g \approx 9.8\ m/sec.^2$

I_{sp} = Specific impulse

It is to be noted that Figure 3.2.1.6 is based on an assumed final orbit perilune altitude of 50 km.

On the basis of the preceding analysis it is concluded that a significant experimental payload capability can be secured with proper mission design and no velocity subsystem modifications. A growth potential exists by upgrading engine performance and/or increasing tankage capacity (contingent on booster upgrading) to achieve greatly increased payload capability. The above capabilities include the potential for injection into circular orbits at low altitude which may be utilized in missions not requiring high electrical power consumption (see Section 3.2.4) and sufficient power capability for active thermal control.

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3.2.2 ATTITUDE CONTROL SUBSYSTEM

Possible attitude control subsystem modifications which must be considered in relation to the scientific experiments fall into the following four categories:

1. Provide capability for continuous orientation of the spacecraft to local vertical in order to perform ground oriented experiments continuously or over long arcs of an orbit.

2. Provide capability for spin stabilization in order to provide vectorial velocity resolution with fewer sensor elements in the experiment.

3. Provide additional attitude maneuver capability for experiment orientation and stabilization.

4. Provide spacecraft stabilization with long boom configurations.

The above modifications, with the possible exception of providing additional attitude control gas capability and resolving the problems associated with boom configurations have to be examined with respect to overall system design and performance characteristics as well as their compatibility with respect to configurations involving multiple experiments with differing operational requirements.

The following paragraphs outline the possible modifications and their impact on system and subsystems performance and experiment compatibility.

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3.2.2.1 Local Vertical Over Limited Arc

The capability of precessing to local vertical can be provided to an approximation over a limited range angle for elliptical orbits or a continuous orbital pass for circular orbits, by an addition of a precision current generator to the closed loop electronics of the flight electronics control assembly. This would provide precession torquing to a preselected gyro, causing the gyro to precess at a fixed rate and the spacecraft to stabilize at the same rate. The wiring provisions for this mode of operation exist at test points in the programmer. An addition of switching capability would be required, in addition to the power supply, if the capability of precessing in either one of the three spacecraft axes is to be provided. Since this precess mode would provide a capability of only one axis at a time it would be necessary to initially maneuver the spacecraft so that the axis around which precession is to take place is normal to the orbital plane. For example, if the roll axis of the spacecraft is chosen as the precession axis, which would be applicable in the case of a polar orbit, the operation mode geometry for a circular orbit would be as shown in Figure 3.2.2.1. Consumption of cold gas for this mode of operation should not differ greatly from cold gas requirements for a normal camera maneuver.

POLAR CIRCULAR ORBIT ROLL PRECESS MODE

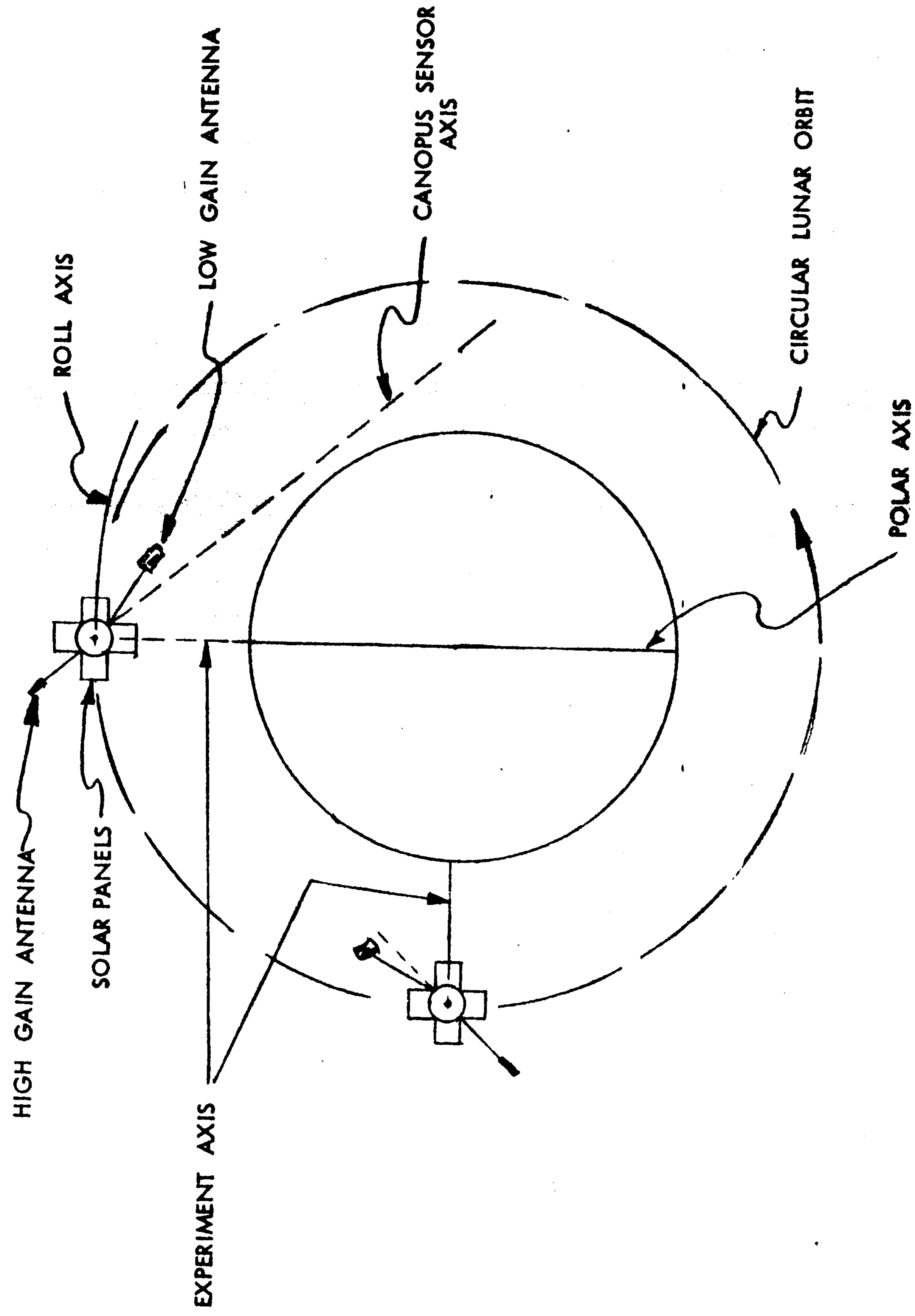

FIGURE 3.2.2.1

3.2.2.2 Local Vertical for Complete Orbit

It should be noted, by reference to Figure 3.2.2.1, that the provision for a continuous precession of the experiment axis to local vertical during the entire lunar orbit results in the following:

1. The Canopus sensor reference axis rotates with respect to inertial space and therefore, inertial reference in this axis would be lost unless extensive gimballing, which would probably require boom mounting, is provided.

2. The solar reference axes would be lost under all conditions, except when the solar ray incidence is normal to the orbit plane, unless multiple switchable sensors or boom mounted gimballed sensors are provided. This is due mainly to the approximately one degree/day precession of the sun with respect to the orbit (Earth's revolution around the Sun) and secondarily due to orbit precession in inertial space.

3. The loss of inertial reference would imply a requirement for feedback by lunar reference sensing, such as a horizon scanner, which would require a sensor development in addition to attitude control subsystem modifications. This would be particularly true for missions of longer duration because of gyro reference drift and generally true for elliptic orbits because of the requirement of torquing at a variable rate due to a varying rate of spacecraft revolution relative to the lunar center of gravity.

3.2.2.2 (continued)

4. Solar panel power output would decrease from its maximum value, at the time of normal solar ray incidence relative to the orbit plane, to approximately P cos (.017 T) where T is the experiment termination time in days, and P maximum power output, unless panel gimballing is provided. For example, without gimballing, at the end of a 30 day mapping mission with the spacecraft continuously operating in the roll precess mode the power output would decrease to less than 75% of the maximum output due to geometry of the problem alone. Compensation for this effect would require either an increased panel area, if the mission is of relatively short duration like 30 days, or a complex panel gimballing arrangement if the mission would be of long duration.

5. The semi-omni antenna coverage nulls would periodically be directed toward the Earth, in the course of the continuous precession, with the exception of the time when the Earth-Moon line is normal or near normal to the orbit plane (perhaps 10-12 days/month). Telemetry data transmission, and perhaps command transmission, would be blocked due to this effect intermittently unless gimballing were to be provided to compensate for the limited coverage under these conditions.

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3.2.2.2 (continued)

6. The directional high gain antenna would require an additional gimbal axis for reasons identical to these outlined in the case of the semi-omni. The requirements for gimballing precision in this case would be more stringent since high directivity is required.

A similar situation exists in the case of continuous precession to local vertical relative to the pitch axis. This would be applicable, for example, in the case of an equatorial orbit as illustrated in Figure 3.2.2.2.. In this case, making the simplifying assumption that the lunar equatorial plane is coincident with the ecliptic plane, no problems relative to the communication subsystem exist. However, the problem with inertial reference and power output of the solar panels is aggravated. In particular, the solar power output varies cyclically, over a single orbital pass, between the maximum and zero even disregarding spacecraft occultation by the moon. Panel gimballing would appear to be the only possible solution to the power output problem in this mode of operation. Inertial reference would have to be replaced by a horizon scanner for reasons identical to those discussed previously.

Orbits with inclination other than polar or equatorial would present a combination of the problem extremes discussed in these cases.

EQUATORIAL CIRCULAR ORBIT
PITCH PRECESSION

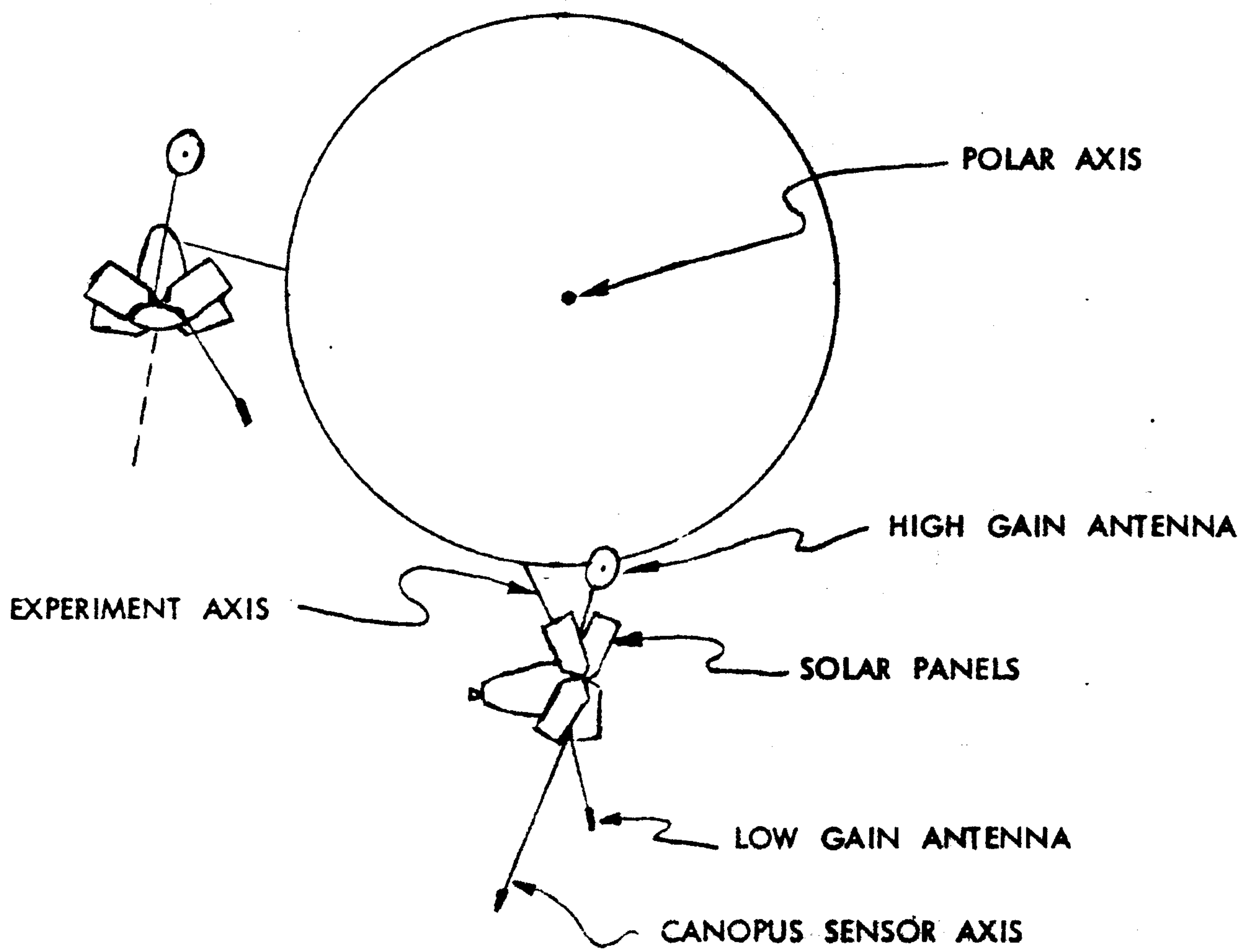

FIGURE 3.2.2.2

3.2.2.3 Spin Stabilization

The case of spin stabilization of the spacecraft presents essentially the same problema as those discussed in connection with torquing to local vertical as well as several additional problems. The additional problems are as follows:

1. No stable reference for the communication, attitude control and surface directed experiments would exist.
2. Solar panel power output would degrade unless the spin stabilization axis is coincident with sun-moon line.
3. Attitude control subsystem maneuver gas budget prior to velocity maneuvers and pre-experiment maneuvers, if necessary, would have to be increased due to increased spacecraft stiffness.
4. Surface related experiments, such as photography or any other range sensitive experiments and/or experiments requiring smear compensation, are not compatible with a spinning spacecraft.
5. Mechanical and structural problems would be associated with spin rates sufficient to stabilize the spacecraft.

A possible minimum modification approach which would partially resolve the above problems would be to take advantage of the available spacecraft capability of slewing, by command, at a rate of 0.5^{o}/sec. in any one of the spacecraft axes. This capability, if used intermittently, could probably provide vector resolution capability with fewer sensors and also provide the capability for scanning the surface in those ground related experiments which

3.2.2.3 (continued)

do not require precision and do not have scanning capability. Continuous slewing should be avoided on the grounds discussed in the preceeding paragraphs. An evaluation of this mode of operation is given in the following subsection.

In order to accommodate some experiments, it may be desirable at some time during the extended mission to rotate the vehicle at a constant rate of $\pm$ 5°/sec. A fuel consumption rate was calculated for this condition using the present vehicle inertias and was found to be .109 or .116 lbs./day depending upon which parts of the sun sensor are in use for a nonspinning vehicle. This compares to a nonspinning value of .012 pounds per day for the present inertia vehicle. Thus the vehicle could be spun up for several days, but not for extensive periods of time without depleting the N_2 gas supply.

A spinning body is stable only when it is spinning about a principal axis of maximum inertia. If it is spinning about an axis of non-maximum inertia, the momentum will transfer to other axes until it is spinning about its axis of maximum inertia. In the Lunar Orbiter the vehicle would be spun up about the sun line-of-sight or the X-axis. This is the axis of minimum inertia and in addition it is not a principal axis. Thus the vehicle is unstable and will try to transfer momentum to its axis of maximum inertia. If Euler's equations are solved under the constraint of a constant .5°/sec. roll rate, the same result is obtained analytically,

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3.2.2.3 (continued)

i.e., the pitch and yaw rates show exponential increase.

Although the vehicle is unstable under the above conditions, the reaction jets have more than enough torque to control the vehicle. Therefore, the control system will have to expend fuel to nullify these inertial coupling torques. These torques are a major factor in extended mission fuel budget and are easily evaluated from Euler's equations. For the pitch and yaw axes they are:

$$T_\theta = |p_o^2 \, I_{xz}|$$

$$T_\psi = |p_o^2 \, I_{xy}|$$

Where po = roll rate in radians/sec.

Ixy, Ixz = inertia products in slug - ft.^{2}

Evaluating: $p_o^2 = .25^{o2}/s^2 = 7.62 \times 10^{-5} \ rad^2/s^2$

$I_{xy} = 1.22$

$I_{xz} = 2.34$

$T_\theta = 178 \times 10^{-6}$ ft. lb.

$T_\psi = 93 \times 10^{-6}$ ft. lb.

For the standard LO configuration, see Weight Report of May 1, 1965

The roll torque is harder to evaluate, but it can be done under several assumptions. If the pitch and yaw torques pin the spacecraft to one switching line, it will operate about that switching line in a one-pulse limit cycle. The pitch and yaw rates can then be approximated by:

$$q, r = (.0025) \sin \omega t \ ^\circ/s$$

3.2.2.3 (continued)

The roll torque from Euler's equation is then represented by:

$$T_\phi = -pq \text{ or } pr\ (I_{xy} - I_{xz}) \sin wt$$

or

$$T_\phi = -1.27 \sin wt \quad \text{ft.-lb.}$$

Expressing T_ϕ as a mean value, $T_\phi = .81\ \mu$ ft.-lb.

For a complete characterization of torques seen by the vehicle, solar pressure torques and gravity gradient torques must be added to the torques derived above. Solar pressure torque is invariant under a rotation about the sun line. Unfortunately the gravity gradient torque is a function of vehicle roll angle. In order to calculate the gravity gradient torque, a computer program that included the effect of a constantly varying roll rate was written. The average value of the torque is dependent on the initial conditions, but will vary about the value calculated for the single set of initial conditions. The calculated average torques are:

$$T_\phi = 1.6 \quad \text{ft.-lb.}$$

$$T_\theta = 3.2 \quad \text{ft.-lb.}$$

$$T_\psi = 1.2 \quad \text{ft.-lb.}$$

The solar pressure torques are:

$$T_\phi = .23 \quad \text{ft.-lb.}$$

$$T_\theta = 1.5 \quad \text{ft.-lb.}$$

$$T_\psi = 5.9 \quad \text{ft.-lb.}$$

3.2.2.3 (continued)

WF_{RAE} = .004 lb./day

The $\Delta\theta$ and $\Delta\psi$ for the coast mode is .13°/second. This results in a fuel usage rate of:

WF_{RAC} = .011 lb/day

The total fuel usage rates are:

WF_{TC} = .116 lb./day, coast mode

WF_{TE} = .109 lb./day, extended mission mode

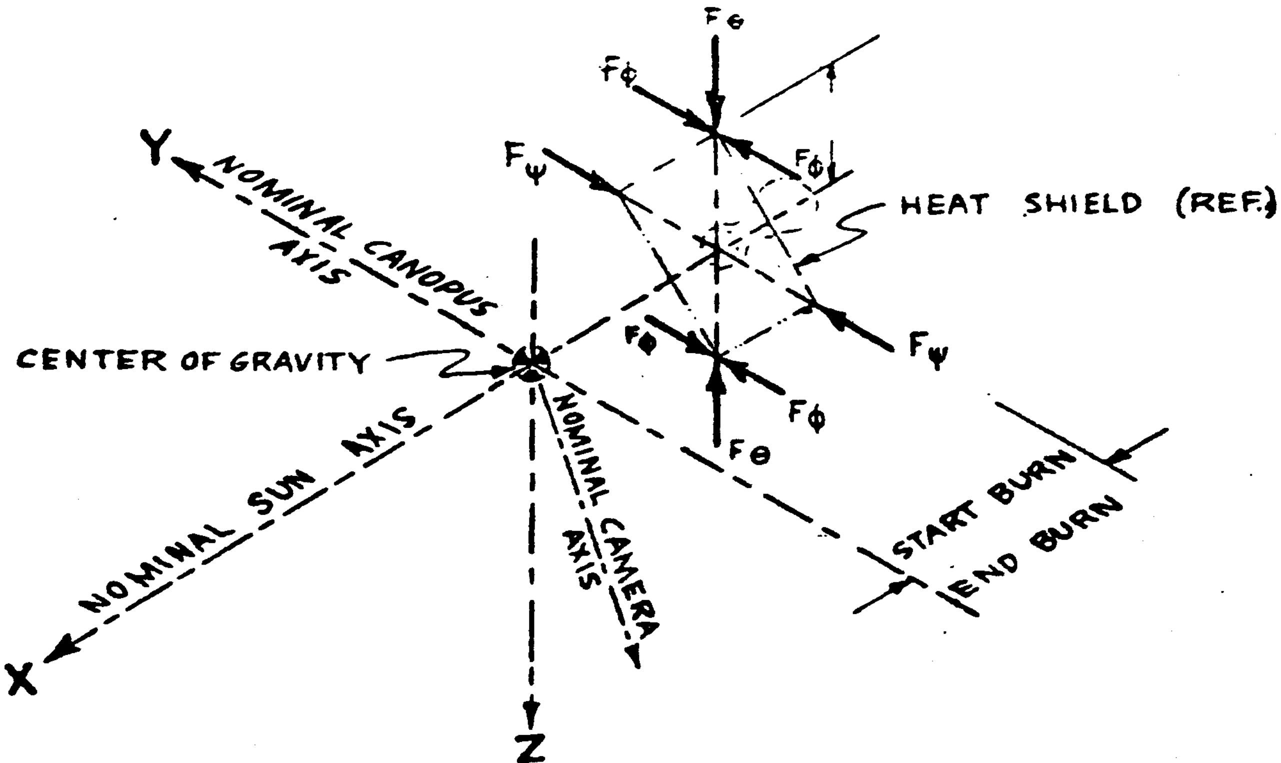
On the basis of the above analysis it is concluded that:

1. The vehicle can be rotated at a .5°/sec. rate, but this increases the fuel consumption rate by a factor of 10.

2. Because of the large increase in fuel usage rate, the vehicle cannot be spun up for extended periods of time without exhausting the nitrogen supply.

3.2.2.4 Control Gas Increase

Attitude control gas budgeting is a function of detailed mission planning, including consideration of maneuver and stabilization requirements. Sample mission plans will be discussed, in connection with specific configurations, in Section 4.0. General preliminary design trade data is summarized in the following paragraphs where the performance numbers are strictly applicable to the current spacecraft configuration. The trade factors are summarized by the relations shown below, as related to Figure 3.2.2.3 and typical



		INERTIA May 1, 1965	REACTION CONTROL			THRUST VECTOR CONTROL	
			Σ JET FORCE	LEVER ARM	ACCELERATION	C.G. TO GIMBAL	ACCELERATION
START BURN	PITCH	123.18	.05 lb	31.37	$.0607°/s^2$		
	ROLL	84.20	.056 lb	17.14	$.0545°/s^2$		
	YAW	134.50	.05 lb	31.42	$.0556°/s^2$		
END BURN	PITCH	76.78	.05 lb	41.79	$.13°/s^2$		
	ROLL	74.54	.056 lb	17.14	$.0615°/s^2$		
	YAW	[illegible]	.05 lb	[illegible]	[illegible]		

3.2.2.3

3.2.2.4 (continued)

maneuver expenditures are tabulated for the basic L. O. configuration.

1. <u>Maneuver</u>

$$W_{fuel\ maneuver} = \frac{\Delta \dot{\Omega} \times \text{Thrust}}{I_{sp}} \quad \text{(for empty vehicle)}$$

$K_\theta \, \Delta \dot{\theta}$ = .006 lb. per degree per second x

$K_\psi \, \Delta \dot{\psi}$ = .007 lb. per degree per second x

$K_\phi \, \Delta \dot{\phi}$ = .014 lb. per degree per second x

	$\Delta \dot{\theta}$		$\Delta \dot{\psi}$	$\Delta \dot{\phi}$	LB N_2		
Close Deadband	.70		.68	.58	.017		
Maneuver	1.32	or	1.36	1.56	.008	or/.01	or /.022
Reverse Maneuver	1.32	or	1.36	1.56	.008	or/.01	or/.022
Open Deadband	0		0	0	0	0	0

Typical maneuvers are:

(a)	Close Deadband and Roll	.039 lb.	
	Reverse Roll	.022 lb.	
	Roll Total	.061 lb.	N_2 roll maneuver
(b)	Close Deadband and Pitch (or Yaw)	.027 lb.	
	Reverse Pitch (or Yaw)	.01 lb.	
	Pitch Total	.037 lb./pitch maneuver	
(c)	Close Deadband and Roll and Pitch	.049 lb.	
	Reverse Pitch and Roll	.032 lb.	
	2-Axis Maneuver Total	.081 lb.	
(d)	Close Deadband and Roll and Pitch and Yaw	.057 lb.	
	Reverse Yaw and Pitch and Roll	.040 lb.	
	3-Axis Maneuver Total	.097 lb.	

(continued)

3.2.2.4 (continued)

(e) Close Deadband and Pitch .027 lb.
Successive Pitch Maneuvers .008 lb.
Re-acquire Sun (1.1°) .007 lb.
.042 lb.

2. <u>Coast</u>

During coast periods the N_2 propellant requirements are the sum of limit cycle, disturbance, and reacquisition of celestial source propellants. For Block I, N_2 use rates are:

$\pm .2^\circ$	Limit Cycle Coast	.07	lb/day
	Reacquisition $\pm .2^\circ$	.022	lb/day
	Disturbance $\frac{.14 \text{ lb. } N_2}{27 \text{ days}}$	.005	lb/day
		.097	lb/day
$\pm 2.0^\circ$	Limit Cycle	.0074	lb/day
	Reacquisition	.0245	lb/day
	Disturbance	.005	lb/day
		.0367	lb/day
	Extended Mission Coast 4.02 lb/335 days	.012	lb/day

Since the above data does not include contingency allowances for changed moment of inertia, cross coupling and flexibility of long booms a conservative budget, including a factor of 2 safety margin, should be used in preliminary estimates. This is summarized in the following tabulation:

Roll maneuver	.12 lbs. of N_2
Pitch or Yaw maneuver	.03 lbs. of N_2
Roll and Pitch	.16 lbs. of N_2
Roll, Pitch and Yaw	.20 lbs. of N_2
Holding $\pm$ 2 Limit Cycle	.20 lbs. of N_2

The extended life nitrogen gas capability of 4.02 lbs. (335 days) can be exchanged for maneuver capability at a rate of decrease of extended life of 84.4 days/lb. If added nitrogen gas tankage

3.2.2.4 (continued)

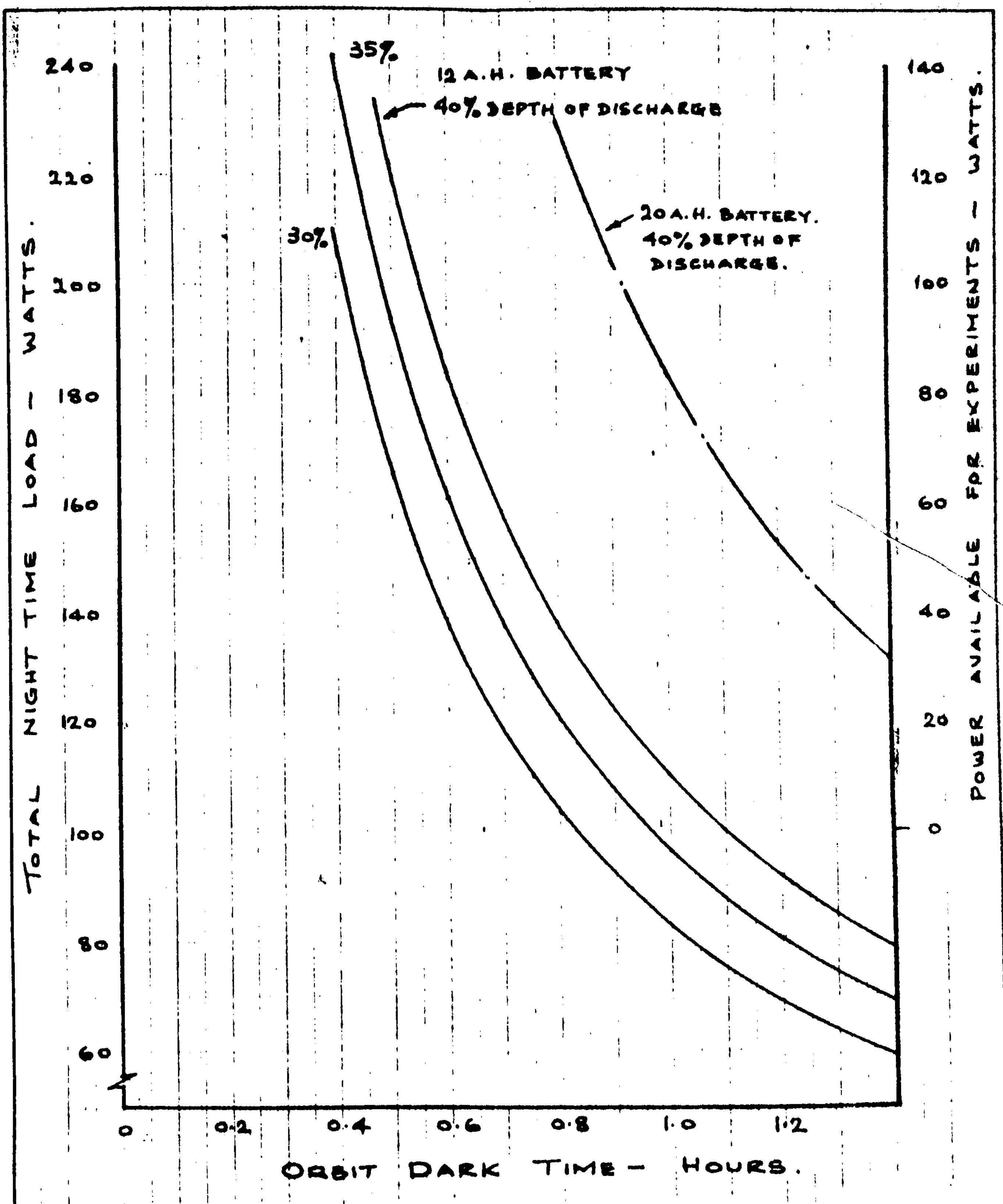
is required then tankage weight can be estimated at 1.6 times the added gas requirements.

3.2.3 POWER SUBSYSTEM

The current design power subsystem performance capability is summarized for night and day operation in Figures 3.2.3.1 and 3.2.3.2 respectively. The performance data of Figure 3.2.3.2 relates to the capability of solar power at the end of the 30 day mission. Since this data includes a degradation factor, which is a function of flight time, the supplementary Figure 3.2.3.3 has to be used to establish day time performance at any time prior to the 30th day.

The availability of power for performing experiments during the time when the spacecraft is occulted by the moon (nighttime) is subject to the constraint that the reliability of the battery subsystem is adversely affected by continual excessive depth of discharge. As far as possible, the recommended depth of discharge should not exceed 40% in lunar orbit.

Under the 40% battery depths of discharge constraint and for the fixed subsystem load of 100 watts, excluding the photographic experiment, it can be seen from Figure 3.2.3.1 that the maximum time in the dark cannot exceed 1.1 hours for the 12 ampere hour battery of the Block I Lunar Orbiter. Additional experiment power requirements could be accommodated by this battery subsystem



	INITIALS	DATE	REV BY INITIALS	DATE	TITLE	MODEL
CALC	J.G.G.	5-14-65			NIGHT-TIME LOAD VS ORBIT DARK TIME, (WITH BATTERY DEPTH OF DISCHARGE AS A PARAMETER)	L.O.
CHECK						
APPD.						
APPD.						

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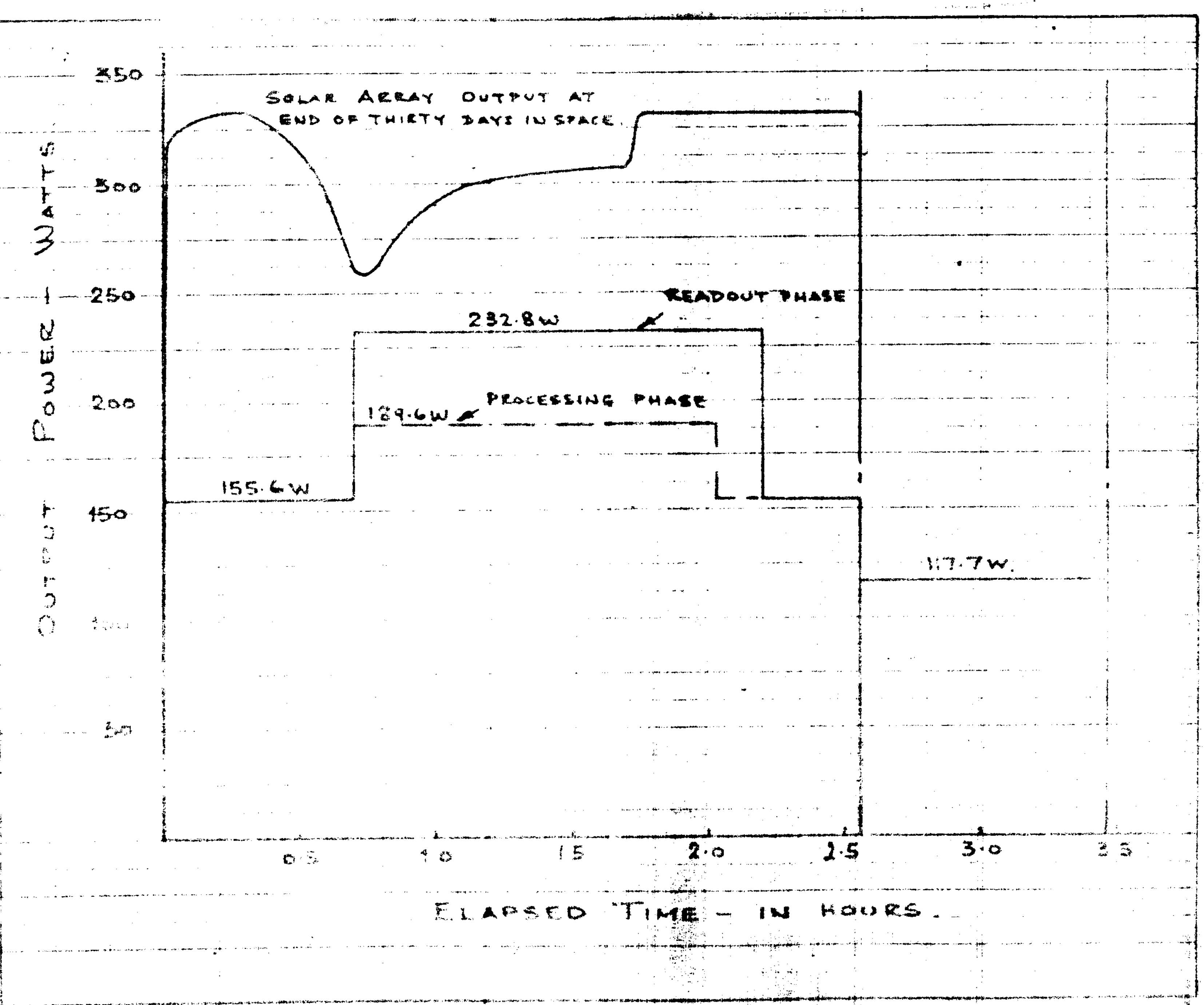
3.2.3.1

REV LTR

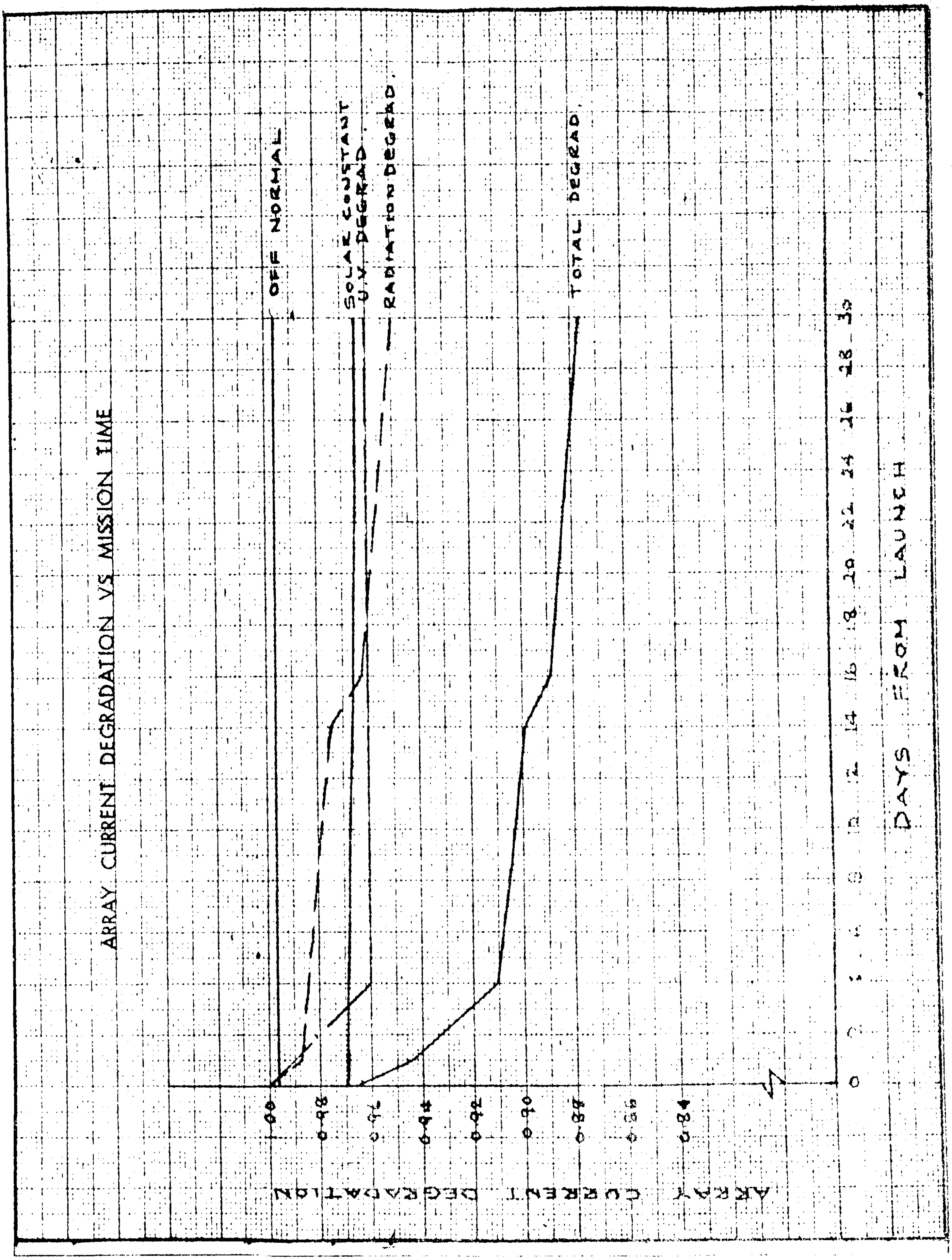


	INITIALS	DATE	REV BY INITIALS	DATE
CALC	J.G.G	7/21/66		
CHECK				
APPD				
APPD				

SUBSYSTEM LOAD PROFILE
LUNAR ORBITS.

MODEL

3.2.3.2



ARRAY CURRENT DEGRADATION VS MISSION TIME
OFF NORMAL
SOLAR CONSTANT
U.V. DEGRAD.
RADIATION DEGRAD.
TOTAL DEGRAD.
1.00
0.98
0.96
0.94
0.92
0.90
0.88
0.86
0.84
ARRAY CURRENT DEGRADATION
0
2
4
6
8
10
12
14
16
18
20
22
24
26
28
30
DAYS FROM LAUNCH

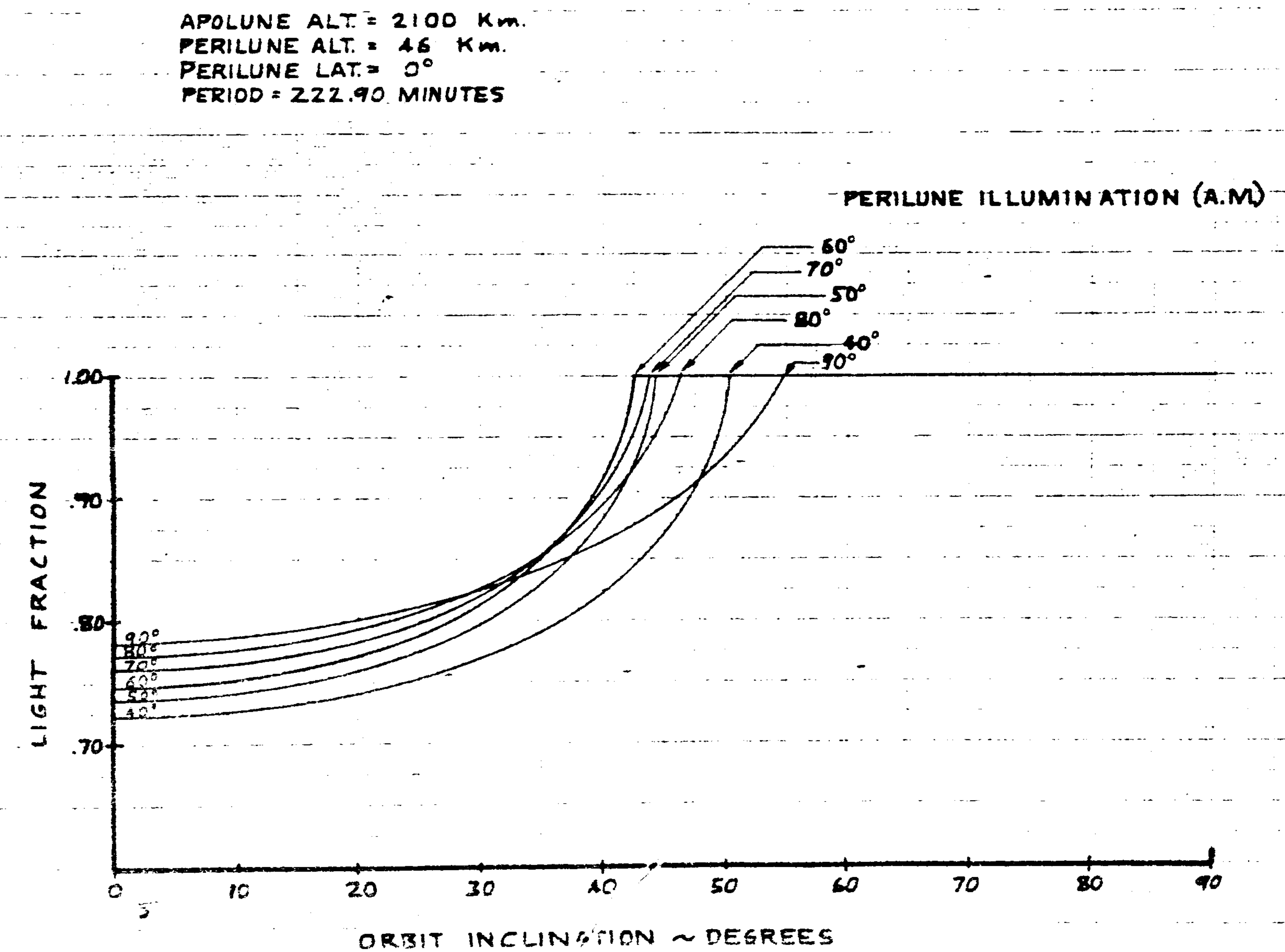
3.2.3.3

3.2.3 (continued)

by either holding the capacity discharge to 110 watt hours (reducing the orbit dark time) or by increasing the risk factor (accepting higher depth of discharge).

The reduction of orbit dark time can be achieved by an increase in orbital inclination and/or variation of illumination at initial perilune. Sample orbital data relating the above factors to the fraction of time the spacecraft remains under solar illumination is given in Figures 3.2.3.4 through 3.2.3.8 which include a variation of orbital apolune, period and latitude of perilune. The data shown in the Figures indicates a preference for high illumination angles at perilune, from the viewpoint of the power subsystem. Since the illumination at perilune is generally a requirement of a surface experiment it cannot be used as a parameter for controlling the fraction of orbit time the spacecraft is under solar illumination. Control of the fraction of orbit time in the dark can, however, be achieved by controlling orbital inclination. This mode of control (i.e. increased orbital inclination) introduces a demand on the experiment field of view or transverse scanning capability, if continuous coverage of large areas is desired (see Figure 3.1.4.1 and 3.1.4.2).

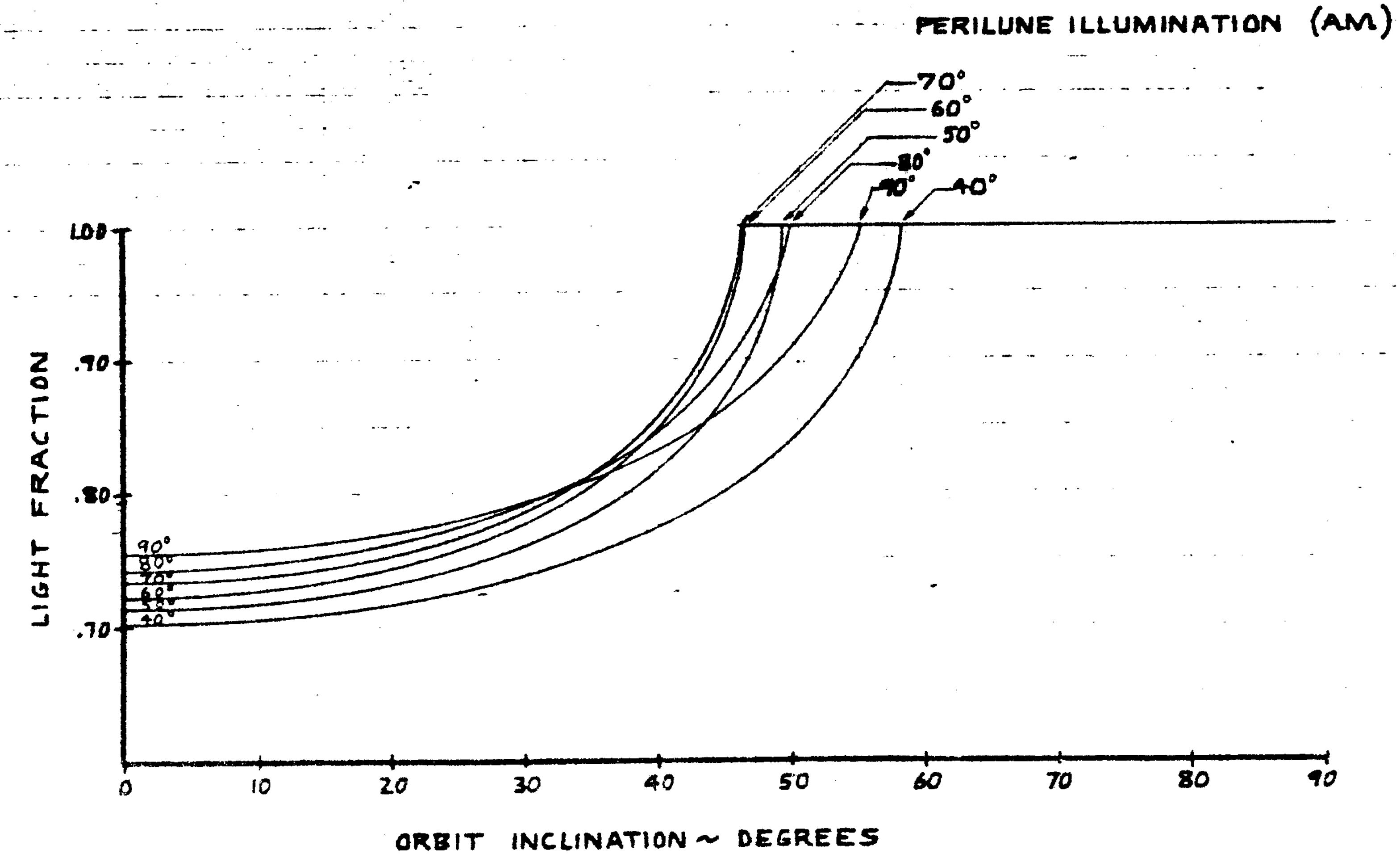
FRACTION OF ORBITAL PERIOD VEHICLE IS IN SUNLIGHT (2100 KM APOLUNE 0° TARGET LAT.)



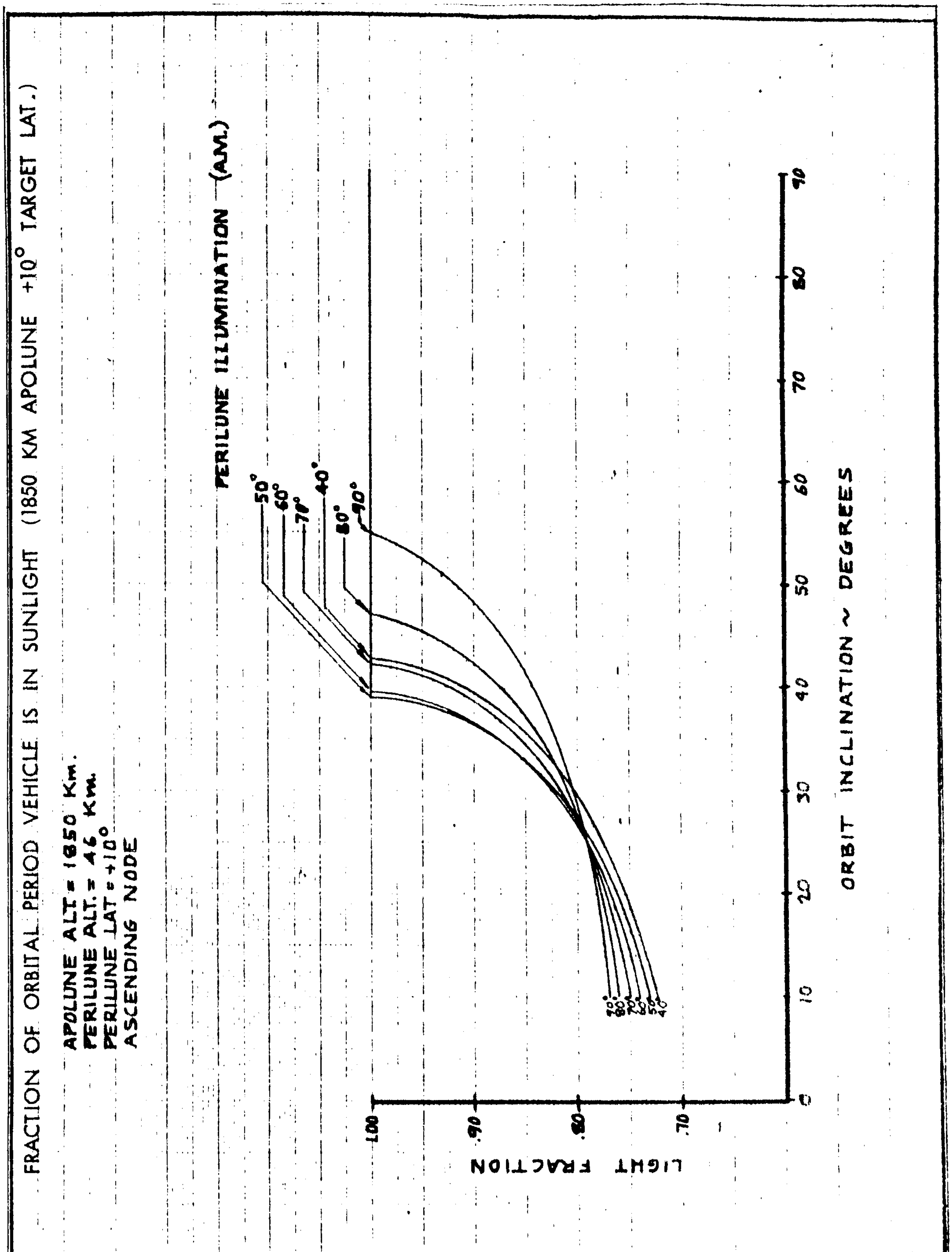
3.2.3.4

FRACTION OF ORBITAL PERIOD VEHICLE IS IN SUNLIGHT (1600 KM APOLUNE 0° TARGET LAT.)

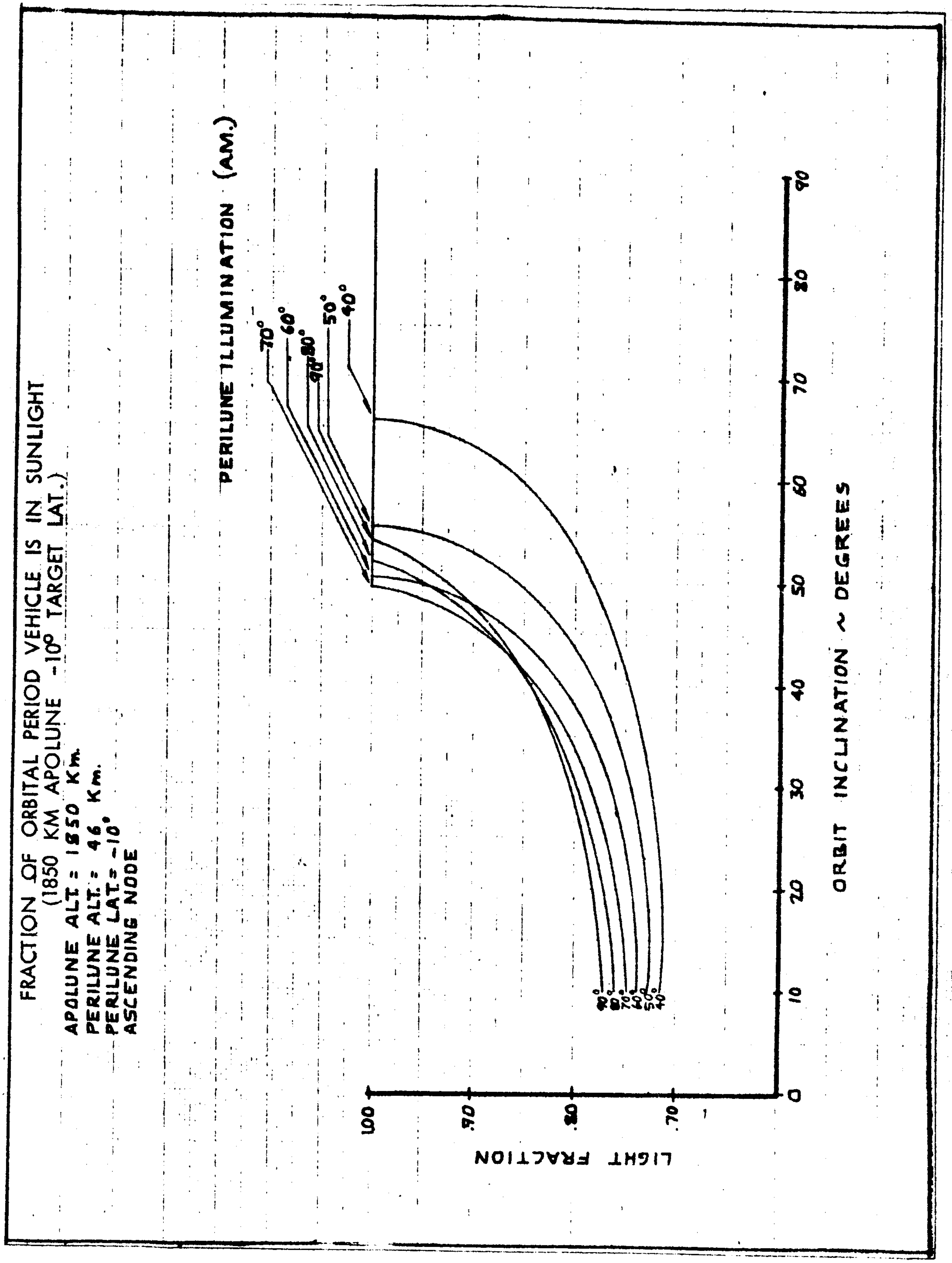
APOLUNE ALT. = 1600 Km.
PERILUNE ALT. = 46 Km.
PERILUNE LAT. = 0°
PERIOD = 193.84 MINUTES



3.2.3.5


FRACTION OF ORBITAL PERIOD VEHICLE IS IN SUNLIGHT (1850 KM APOLUNE +10° TARGET LAT.)
APOLUNE ALT = 1850 Km.
PERILUNE ALT. = 46 Km.
PERILUNE LAT = +10°
ASCENDING NODE
PERILUNE ILLUMINATION (AM.)
50°
60°
70°
40°
80°
90°
LIGHT FRACTION
1.00
.90
.80
.70
ORBIT INCLINATION ~ DEGREES
0
10
20
30
40
50
60
70
80
90

3.2.3.6



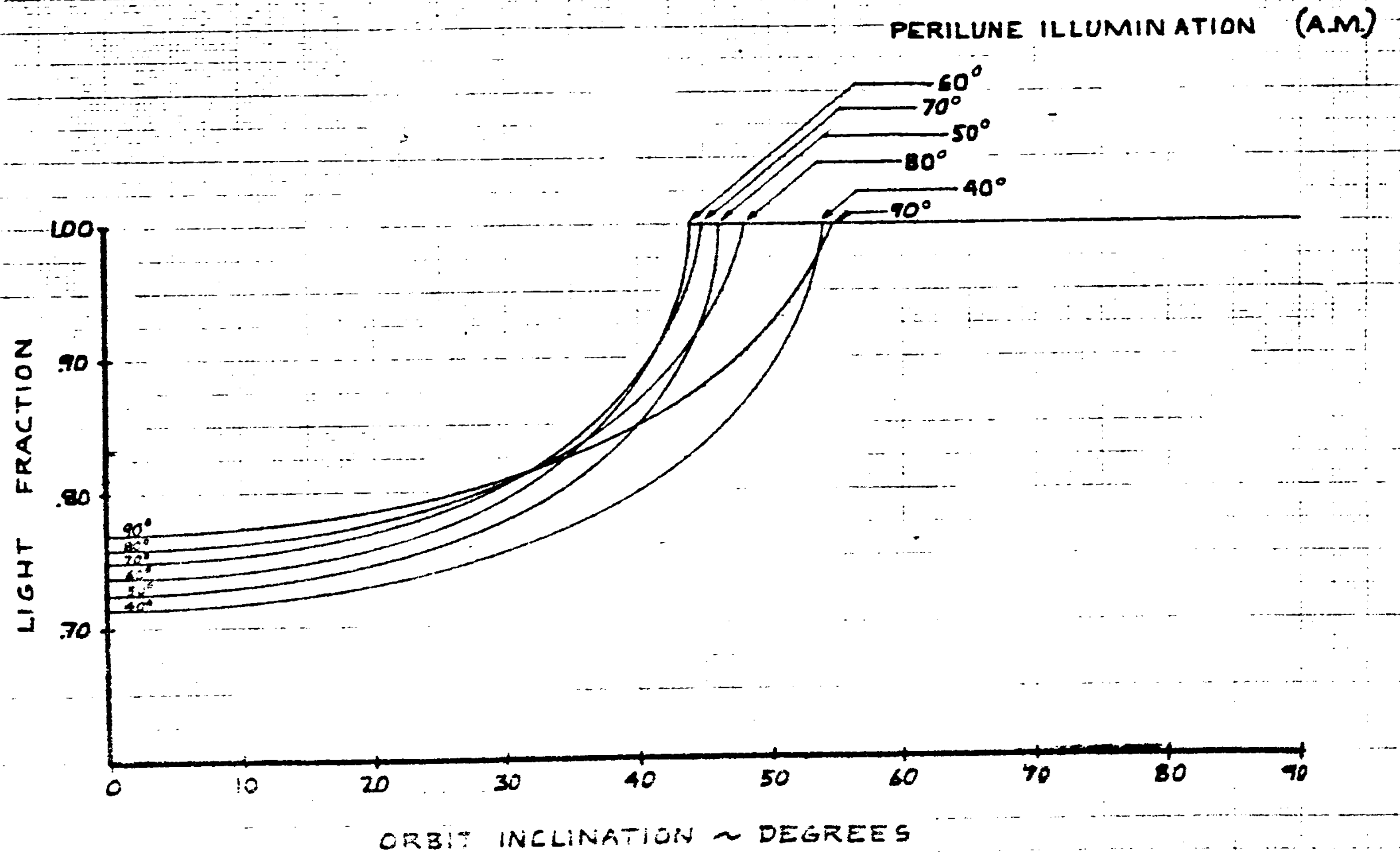
FRACTION OF ORBITAL PERIOD VEHICLE IS IN SUNLIGHT
(1850 KM APOLUNE -10° TARGET LAT.)

3.2.3.7

USE FOR DRAWING AND HANDPRINTING — NO TYPEWRITTEN MATERIAL

FRACTION OF ORBITAL PERIOD VEHICLE IS IN SUNLIGHT (1850 KM APOLUNE 0° TARGET LAT.)

APOLUNE ALT. = 1850 Km.
PERILUNE ALT. = 46 Km.
PERILUNE LAT = 0°
PERIOD = 208.20 MINUTES



3.2.3 (continued)

The effectiveness of controlling "night" time by inclination variation can be shown by cross-referencing Figures 3.2.3.4 and 3.2.3.1 for two assumed inclinations of, say, 20^{o} and 35^{o} and a given 60^{o} illumination at perilune. The corresponding light fractions from Figure 3.2.3.4 are .77 and .85 yielding .875 hrs. and .55 hrs. of night time respectively for the given period of 3.71 hours. The available power for experiments corresponding to these cases is, by reference to Figure 3.2.3.1 40% discharge curve, 26 watts and 100 watts respectively.

The alternative to the above would be to provide additional battery capacity which would involve development and qualification testing as well as additional spacecraft inert weight. The capability of a 20 amp. hour battery, involving an inert weight penalty of approximately 20 lbs., is shown in Figure 3.2.3.1. This modification does not provide a significantly greater capability than that attainable with moderate increases in orbital inclination inasmuch as nighttime operation of experiments is concerned. However, increased battery capacity is of significant importance relative to the capability of spacecraft to approach low altitude circular lunar orbits. This arises due to the constraint that battery charge rates should be held below 1/4 of the battery capacity to prevent cell pressure build-up and degradation.

USE FOR TYPEWRITTEN MATERIAL ONLY

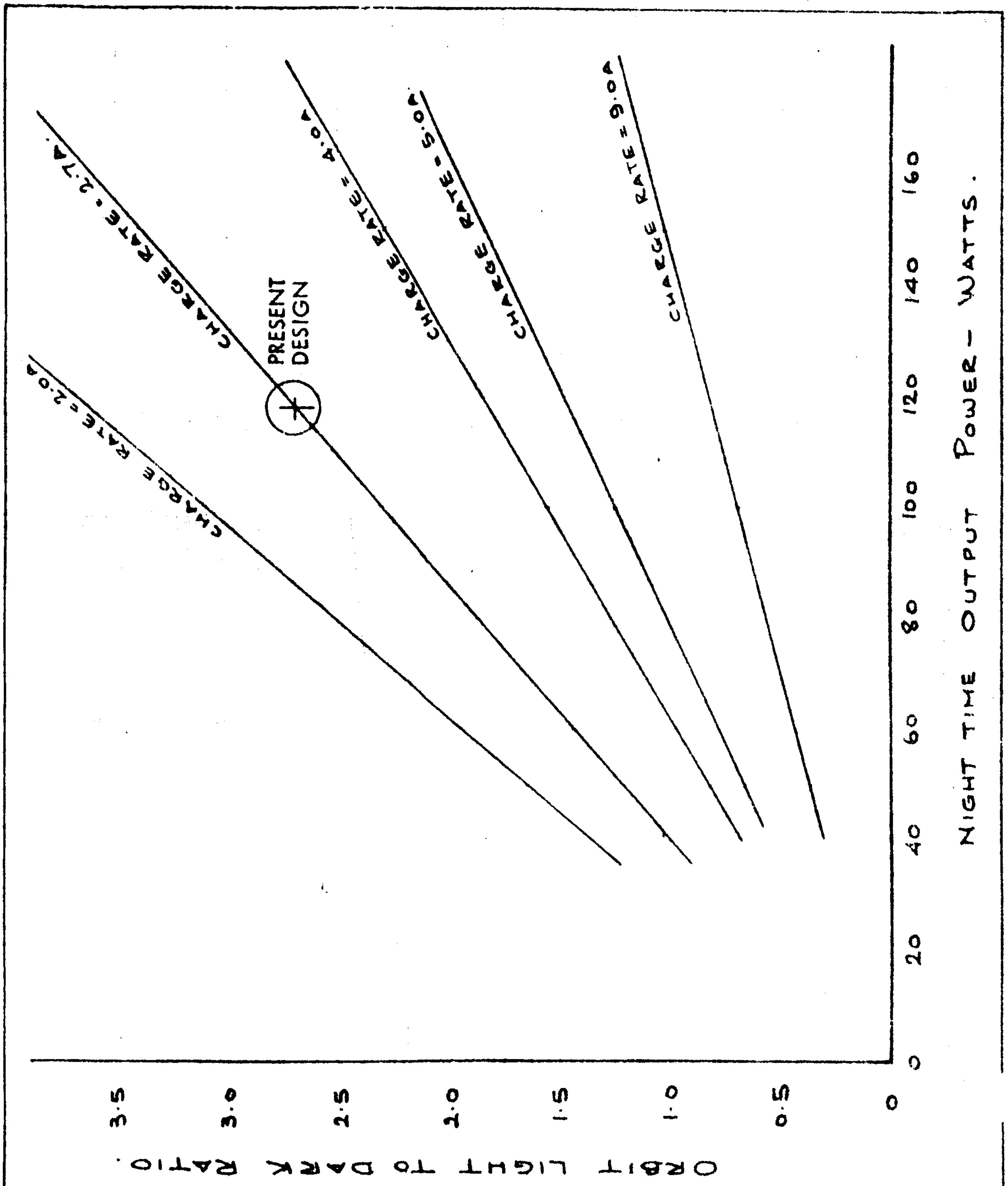
3.2.3 (continued)

The above constraint can be summarized in terms of the relation of the light to dark ratio of the orbit to night time load for a given maximum charge rate constraint. This is summarized in Figure 3.2.3.9 with the charge rate as a parameter. Minimum battery capacity for a given light to dark ratio and a given nighttime load can be obtained from the figure using the relation:

Min. Capacity = 4 x Min. Charge Rate

For example, a near zero altitude circular orbit would have a light to dark ratio of approximately one. Assuming a nighttime load of 150 watts the min. required charge rate would be 9 amps. and the corresponding battery capacity would have to be 36 amps. hrs.

Conversely, given a 2.7 amp. charge rate, corresponding to a 12 amp. hr. battery , and near zero altitude circular orbit, the maximum nighttime power load would have to be limited to 40 watts. On the basis of the above observations with respect to Figure 3.2.3.9 and the consideration that power output during nighttime will generally run in excess of 140 watts, when experiment, thermal control and data recording requirements are included, it is self-evident that with the existing battery subsystem a minimum light to dark ratio of approximately 3.2/1 would have to be achieved by proper orbit design. Capability for sustaining the above power load in a low altitude circular orbit could be



	INITIALS	DATE	REV BY INITIALS	DATE	TITLE	MODEL
CALC	J.G.G	7/15/65			NIGHT-TIME OUTPUT POWER VS LIGHT TO DARK RATIO, FOR A GIVEN BATTERY CHARGE RATE.	
CHECK						
APPD						
APPD						

3.2.3 (continued)

achieved by retrofitting the spacecraft with four parallel 12 amp. hr. batteries, at a cost of an additional 90 lbs. of battery weight, or an approximately 45 amp. hr. battery at a somewhat lesser cost in inert weight but involving development. The possibility of a drastic reduction in power load is questionable since a fixed load of 100 watts exists during the dark time regardless of experiment requirements. The daylight power loads would also become adversely affected by a decrease of the light to dark ratio. For the 1:1 light to dark ratio, a requirement would exist for an approximately 12 ampere charge rate which could not be supported with the current output of the solar panels, as can be seen by reference to Figures 3.2.3.2 and 3.2.3.3, when the other power requirements are taken into consideration.

In order to support the above charge rate in conjunction with the video transmission mode, in those configurations including the photographic subsystem, an increase in panel area and weight to roughly 97 ft^2 and 100 lbs. respectively would be required. In the absence of the photographic subsystem the above area can be reduced approximately to 85 ft^2 and 87 lbs. if a requirement for the high gain communications subsystem exists and to 77 ft^2 and 79 lbs. if no requirement for the high gain communication subsystem exists.

3.2.3 (continued)

Furthermore, due to the decrease in power output resulting from a rise in solar temperature at the time when their backs are exposed to surface reflected radiation (Figure 3.2.3.1) an impairment of capability of transmitting video data, in the configurations including the photographic experiment, would occur in the case of low altitude circular orbits. This would effectively limit the transmission capability to half a frame/orbit and result in transmission times of up to 33 days, compared to the current 9.5 days, for the total possible number of photographic frames.

On the basis of the above considerations it appears that low circular orbits are not advisable from the power subsystem point of view.

The preceeding conclusion reinforces the conclusion reached in conjunction with consideration of the trades available relative to velocity control subsystem in section 3.2.1 that in the interest of minimization of system modifications and maximization of experimental payload capability lunar orbits of high eccentricity are preferable.

Power subsystem output will be reduced during "daytime" operation due to the misalignment of the normal to solar panel plane with the solar illumination axis in the course of alignment of the spacecraft to the attitude appropriate to the experiment. Assuming that

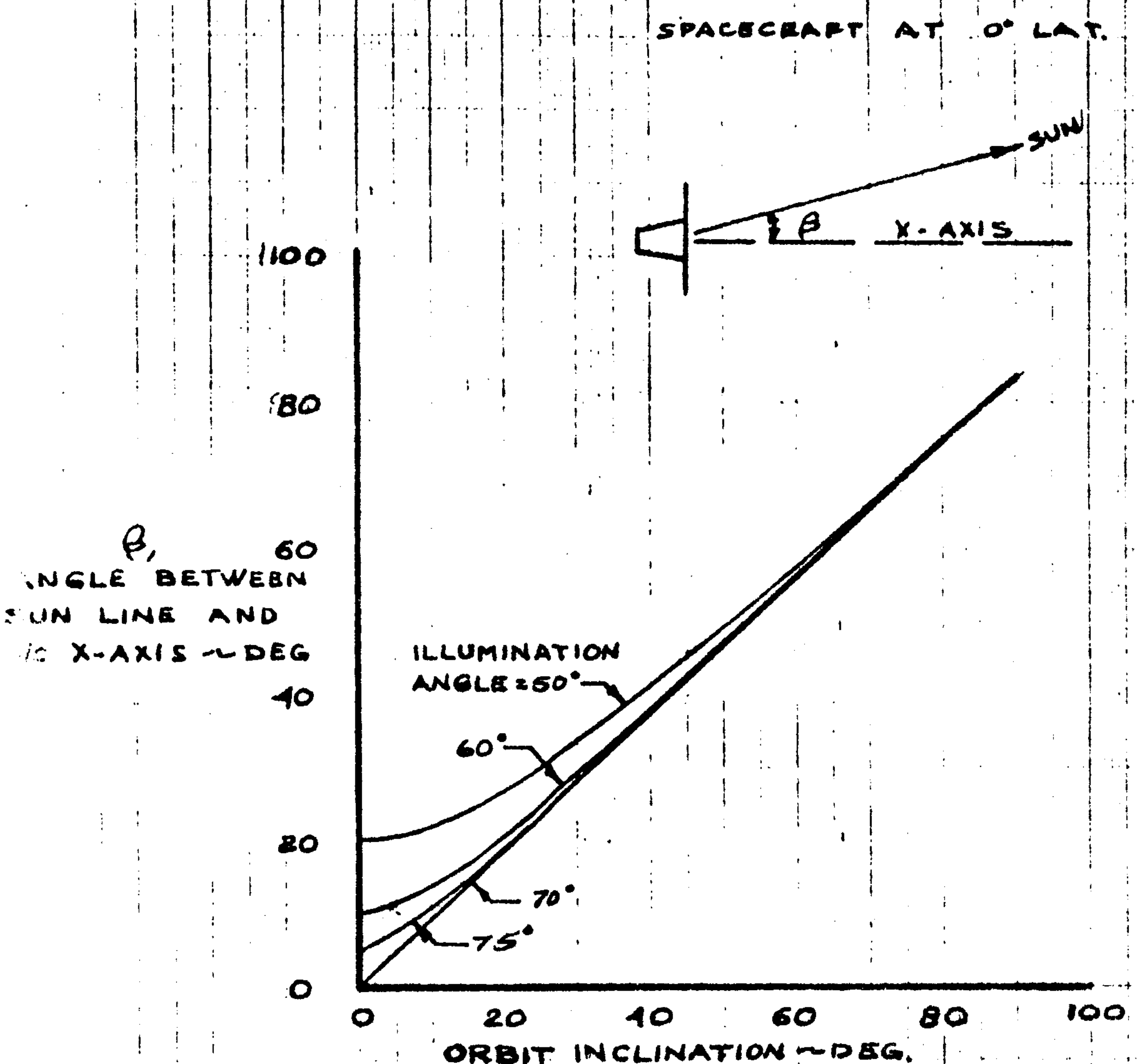
3.2.3 (continued)

spacecraft realignment into the plane of the orbit and to local vertical is required by the experiment, the solar power output decrease can be estimated as being proportional to the cosine of the misalignment angle. The misalignment angle as a function of orbital inclination and with the illumination at the subspacecraft point on the lunar surface is shown in Figure 3.2.3.10 for a sample case of the spacecraft at the equator. The same relation holds with respect to heat input to the equipment mounting deck.

The foregoing discussion does not take into account eclipses. Data on lunar eclipses in the 1965-1970 time period are in Table 3.2.3-1. Additional storage battery capacity or other means of storing heat energy may be required if high probability of operation through the eclipses is an experimental requirement.

USE FOR TYPEWRITTEN MATERIAL ONLY

SOLAR RAY INCIDENCE ON SOLAR PANELS DURING PHOTOGRAPHIC MODE



3.2.3.10

TABLE 3.2.3-1

LUNAR ECLIPSES 1965 - 1970

Date	Type	Duration: Time from Entering Penumbra to Exiting	Duration: Time from Entering Umbra to Exiting	Duration: Total Eclipse Time
Dec. 8, 1965	Penumbral	4 hr 4.8 min	0	0
May 4, 1966	Penumbral	4 hr 10.1 min	0	0
Oct. 29, 1966	Penumbral	4 hr 38.2 min	0	0
April 24, 1967	Total	5 hr 16.3 min	3 hr 23.5 min (3 hr 34 min)	1 hr 18.3 min (1 hr 22 min)
Oct. 18, 1967	Total	6 hr 10.7 min	3 hr 39.5 min (3 hr 26 min)	1 hr 0.7 min (50 min)
April 13, 1968	Total	Not Available	(3 hr 26 min)	([illegible] min)
Oct. 6, 1968	Total	Not Available	(3 hr 28 min)	(1 hr 2 min)
April 2, 1969 (Estimated)	Penumbral	Not Available	0	0
August 27, 1969 (Estimated)	Penumbral	Not Available	0	0
February 21, 1970	Partial	Not Available	(52 min)	0
August 17, 1970	Partial	Not Available	(2 hr [illegible] min)	0

Reference: "Astronomical Phenomena 1965, 196[illegible], 196[illegible]"

"Canon of Eclipses" 1887 (for data in parentheses)

3.2.4 PROGRAMMER

The spacecraft programmer capacity is sufficient to accommodate additional control functions with a perhaps increased frequency of command reprogramming requirements. The modification to the programmer cannot be determined until after a detailed analysis of the number and type of functions to be controlled. In general terms it can be stated only that, with the possible exception of the case when the photographic subsystem is deleted, additional output switching and an associated output matrix modification would be required. In any case wiring modifications will be required.

3.2.5 COMMUNICATIONS SUBSYSTEM

The Block I Lunar Orbiter Communication system is designed to operate in three modulation modes (See Volume II, D2-100369-2)

Mode 1 - PN ranging and/or telemetry

Mode 2 - Video and telemetry

Mode 3 - Telemetry

Each of these modes comprise different input signals, data rates, bandwidths, and output antenna and power configurations.

Mode 1 : Ranging and/or telemetry

Telemetry data rate	- 50 bps
Telemetry subcarrier frequency	- 30 Kc
PN Ranging data rate	- 500 Kbps
Ranging subcarrier	- None, PN code directly modulated on carrier
Transmitted bandwidth	- 3-1/3 Mc
Output power and antenna	- 400 mw and omni-antenna

USE FOR TYPEWRITTEN MATERIAL ONLY

3.2.5 (Continued)

Mode 2 : Video and telemetry

Telemetry data rate	- 50 bps
Telemetry subcarrier frequency	- 30 Kc
Video bandwidth	- 230 Kc
Video subcarrier frequency	- 310 Kc (vestigal sideband)
Transmitted bandwidth	- 3-1/3 Mc
Output power	- 10 watts (TWTA)
Antenna	- 23.5 db high gain

The wide bandwidth of the video signal requires the use of the TWTA (traveling wave tube amplifier) and high gain antenna for this mode. Although the data rate of the ranging system is even larger (500 Kc vs. 230 Kc), the effective bandwidth of the ranging signal is only a few cycles as correlation detection is used at the DSIF.

Mode 3 : Telemetry only

Telemetry data rate	- 50 bps
Telemetry subcarrier frequency	- 30 Kc
Output power	- 400 mw
Antenna	- Omni low gain

Mode 3 is intended for the transmission of narrow bandwidth, low power, low gain antenna telemetry data. A separate 30 Kc subcarrier oscillator (for redundancy) is used for Mode 2 and for Modes 1 or 3.

Several modifications, providing a wide range of transmission capability, can be considered under the ground rules of minimum modifications.

3.2.5 (Continued)

These modifications, arranged in order of increasing capability are tabulated below in terms of transmitter and antenna combinations:

Max. Rate (bps)	Transmitter	Antenna
1000	10 Watt	Low Gain
3000	0.5 Watt	High Gain
230,000	10 Watt	High Gain

The subsystem modifications associated with the above tabulation are discussed in the following paragraphs.

The combination of 10 watt transmitter and high gain antenna does not require a direct modification of the communication subsystem, under the assumption that no signal conditioning is required. The only modification would consist of a provision for energizing either the video readout or the tape recorder storage readout of the other experiments. Video and other experimental data would be transmitted on a time share basis with intermediate tape recorder storage provided to the other experiemtns. The capability of transmitting up to 3000 bps, using the combination of half watt transmitter and high gain antenna, would require a modification of the multiplexer encoder in order to combine the transmission of spacecraft performance data and experimental data. Additionally, the telemetry subcarrier and band pass filter would have to be changed from the present 30 kc frequency to 20 kc. This change is required to prevent interference with the video data. A provision for selecting the high gain antenna in combination with the low power transmitter would have to be provided if the high data rate capabilities necessary for either video or X-ray

USE FOR TYPEWRITTEN MATERIAL ONLY

3.2.5 (Continued)

data transmission on the same mission were to be preserved. Switching back to the current normal modes 1 and 2 would also be necessary in this case.

The capability to transmit up to 1000 bps, using the combination of low gain antenna and the high power transmitter would require a modification of the multiplexer encoder, a modification of the subcarrier oscillator (30 kc to 20 kc) and a provision for selecting the above combination of transmitter and antenna.

It is to be noted that in either of the two latter cases the preservation of high data rate capability would be contingent on the capability to switch back to normal mode 1 and 2 operation on a time share basis with these modes. This does not appear to have any advantage over the high data rate system, utilizing the high gain antenna and high power transmitter and should be considered only if the total data rates are sufficiently low. For those experiment configurations where total data rates are lower than 1000 bps it may be advantageous to eliminate the high gain antenna and low power transmitter and utilize the weight (approximately 15 lbs.) for additional payload. Similarly, the elimination of the TWTA and the low gain antenna (approximately 12 lbs. payload gain) may be justified if the total data rates can be held below 3000 bps. In either of the above cases no switching provisions would be required.

On the basis of the experiment data rates, provided by L-5382 Statement of Work and the projected payload capability of the Lunar Orbiter it

3.2.5 (Continued)

appears unlikely that the total data rates could be held below 3000 bps. This is self-evident in the cases I - III (Photo System retained), where video transmission capability has to be preserved, and will become subsequently apparent for Case IV (no Photo System) since this configuration is capable of supporting the whole experiment complement with a total data rate requirement of approximately 60,000 bps.

In view of the above observations is appears desirable to preserve the current spacecraft communications subsystem in an unchanged form, except for the provision of input switching to the video system, and to provide storage capability for the experiment complement which would be sufficient for:

1. Experiment data storage during the time experiments are activated.
2. Experiment data storage during sun and/or earth occultation of the spacecraft by the moon.

Data transmission, utilizing the high gain antenna and 10 watt transmitter, would be initiated on a time share basis with the video, whenever video transmission is required, at a time when the limit of storage capacity is approached. Transmission would take place at the maximum rate, consistent with tape recorder readout capability, in order to minimize transmission time.

Space qualified tape recorders meeting the requirements of experiment groupings supportable by Lunar Orbiter are available on the market and their specifications will be discussed in connection with specific configurations in Section 4.0.

3.2.6 PHOTOGRAPHIC SUBSYSTEM

The extent of photographic subsystem modifications associated with the four Cases as specified by the L-5386 Statement of Work, is discussed in the following paragraphs.

Case I, requiring the preservation of the total photo subsystem capability, does not require any modification evaluation.

Case II requiring the removal of the high resolution portion of the photographic subsystem, involves several modifications and results in weight and power savings as shown in the table below, in terms of individual items deleted.

Item	Weight	Power
1. 24" lens	13.32#	--
2. 24" Lens Window Assembly (Window heater)	.30#	3.5 Watts
3. Folding mirror	.75#	--
4. 24" Platten and Actuator Assembly	1.70#	--
5. 24" Shutter Assembly	1.70#	--
6. Light Baffling (24" lens only)	.30#	--
7. 24" Window	.48#	--
8. V/H Sensor		
Mechanism*	5.54#	10.5 Watts (when on)
Electronics	3.94#	
TOTAL	27.99#	14. Watts

* Complete removal of the V/H Sensor is contingent on increasing the 3" lens aperture, increasing shutter speed and accepting a resoltuion degradation.

USE FOR TYPEWRITTEN MATERIAL ONLY

3.2.6 (Continued)

The potential volume savings accrued by the above deletions are shown in Figures 3.2.6.1 through 3.2.6.5.

Because of the complexity, from the viewpoint of mechanical, thermal and subsystems, of integrating interchangeable experimental packages within the pressure envelope of the photographic subsystem the utilization of this volume for experiments is not recommended.

The modification required concurrently with the removal of the items tabulated above and the impact of these modifications on photographic subsystem capability are discussed below:

The film metering roller encoder should be modified to advance 2.874" of film rather than 11.732" in order to fully utilize the film capacity. This modification will result in a capability of 793 frames of moderate resolution photographs. This capability would result in the potential of mapping an area of 25° x 360° with stereo coverage to a nominal resolution of 32 meters.

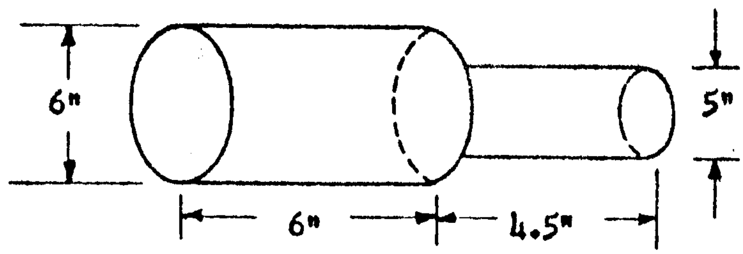
The film rollers adjacent to the 24" platten should be removed to insure proper film transport.

Cabling changes will be required to provide an Image Motion Compensation (IMC) drive signal by command from the programmer unless shutter speed is increased and a somewhat lower resolution (9-18 meters) is accepted. An A/D converter, within the envelope of the present V/H mechanism will have to be provided in this case. The weight statement in this case would result in a decreased saving of five pounds.

USE FOR TYPEWRITTEN MATERIAL ONLY

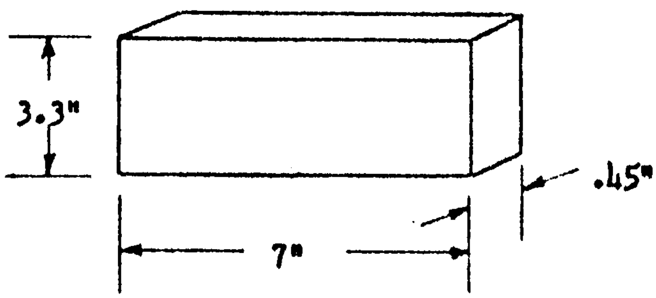
Space Available

1) 24" Lens Assembly



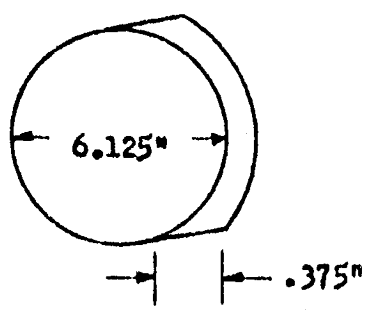
Volume = 258 cubic inches
includes light baffle

2) Folding Mirror



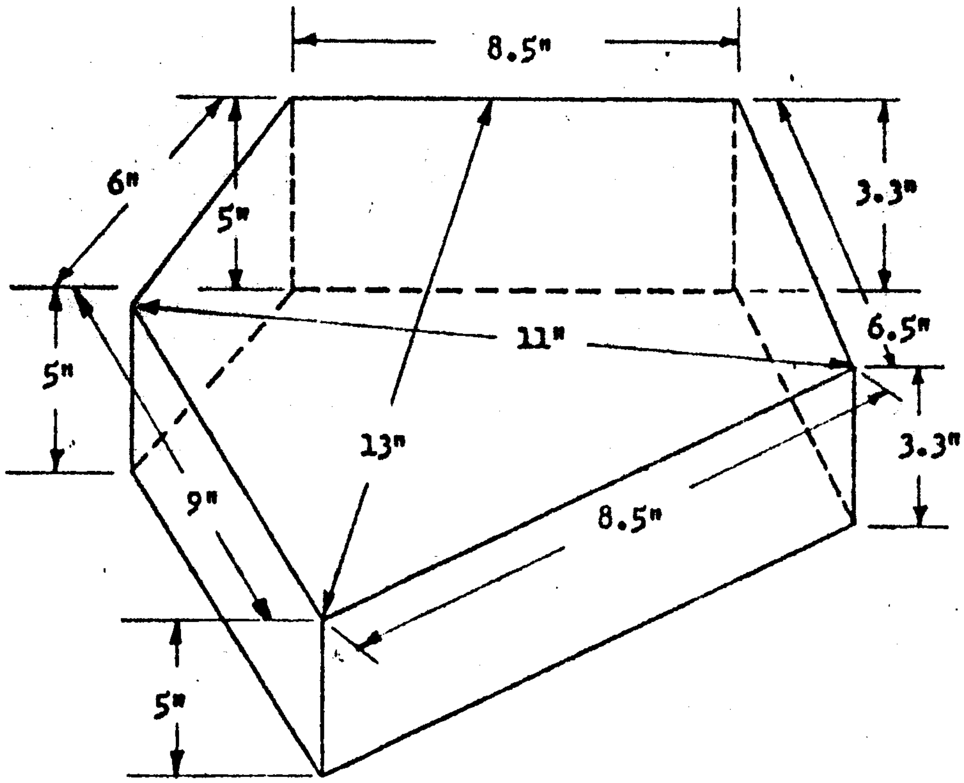
Volume = 10 cubic inches

3) 24" Window



Volume = 11 cubic inches

4) Enclosed light path



Volume = 405 cubic inches

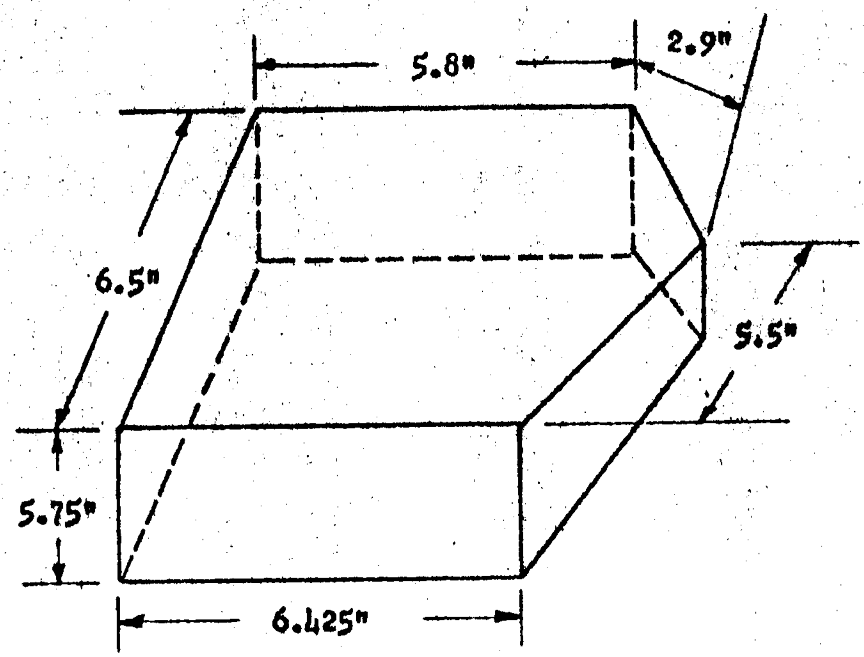
includes 24" shutter assembly

light path baffling

USE FOR TYPEWRITTEN MATERIAL ONLY

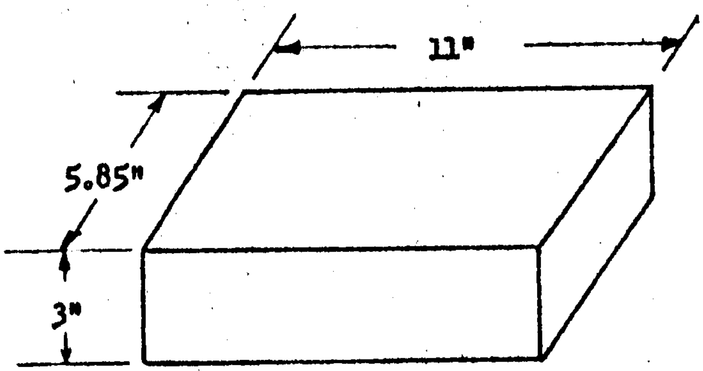
5) V/H Sensor

Mechanism



Volume = 246.8 cubic inches

Electronics



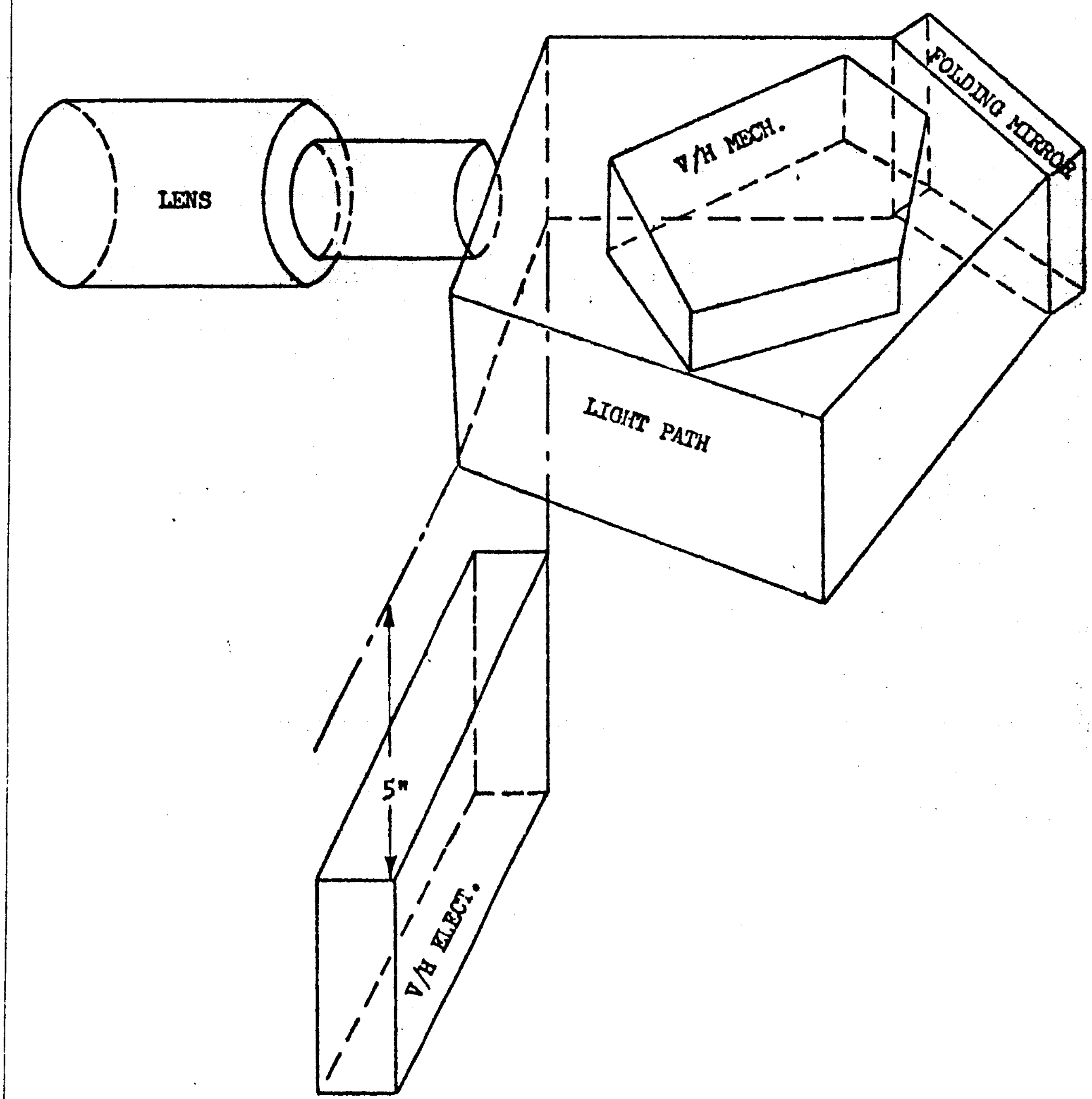
Volume = 194 cubic inches

Total volume Case II = 1220 cubic inches

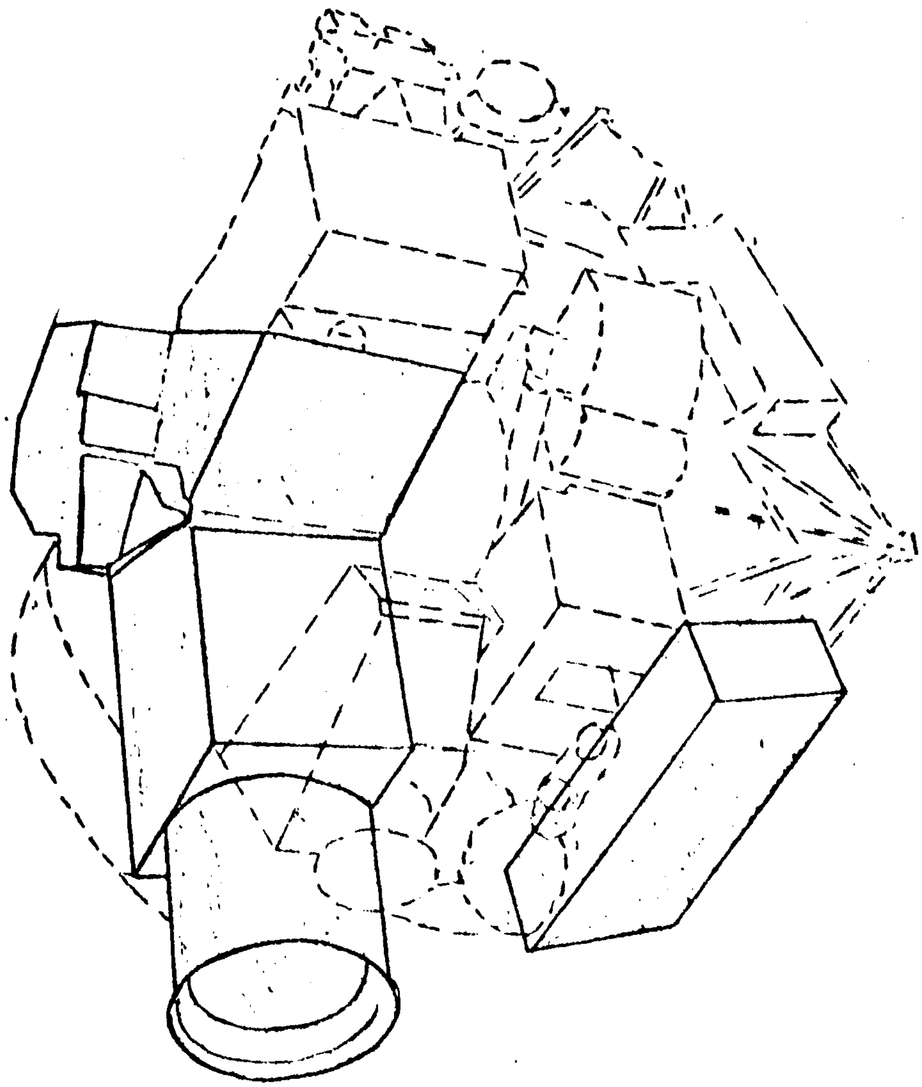
Note: Angular shape may allow increase of this volume by 25%.

USE FOR TYPEWRITTEN MATERIAL ONLY

Combined component spacing and relationship



3.2.6.4



3.2.6.5

3.2.6 (Continued)

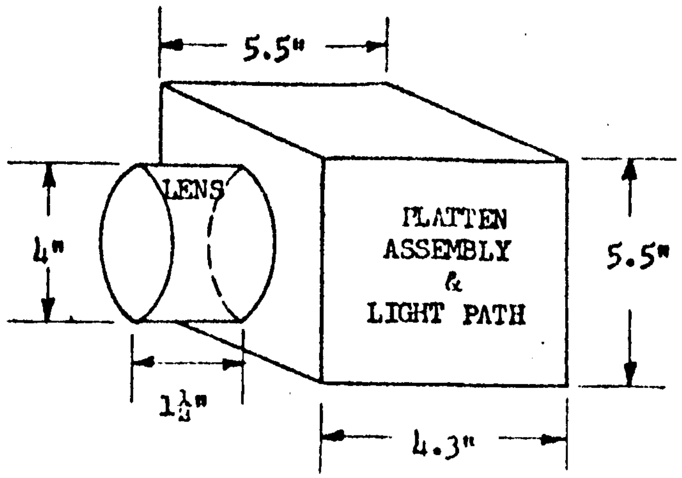
Alternately, the IMC function can be eliminated if a degradation in resolution is acceptable, as may be the case with high altitude (200 km) mapping type missions yielding 40 meter resolution. The modification, in this case, would be to take advantage of the full capability of the 3" lens (F2.8) instead of stopping it down to its present mode of f5.6. This modification would allow the shutter speed to be increased to 1/100, 1/200 and 1/400 seconds thereby limiting smear to 18 meters at the slowest shutter speed corresponding to low light levels occurring at illumination angles in the neighborhood of 75°. Smear would be limited to 4.5 meters in the neighborhood of 50° illumination. The 24" window area must be sealed.

Case III, requiring the removal of the low resolution lens, results in minor weight savings of 2.90 lbs. consisting of 2.6 lbs. of lens and platten actuator and .030 lbs. of window. Now power budget decrease is achieved.

The modification associated with this deletion involve a reattachment of the shutter exposure setting mechanism (presently attached to 3" camera), sealing of the 3" window and modification of the film metering mechanism would result in the film metering roller encoder advancing 8.936" of film rather than 11.732". This would provide full utilization of the film capability and result in an increase of capacity from 194 frames of high resolution photos to 255 frames.

The volume saving associated with this modification is shown in Figure 3.2.6.6 and its usage for other experiments is not recommended on the same grounds as discussed in Case II.

Case III Low Resolution Camera Removal



Volume = 149 cubic inches

USE FOR TYPEWRITTEN MATERIAL ONLY

3.2.6.6

USE FOR TYPEWRITTEN MATERIAL ONLY

3.2.6 (Continued)

In general, this modification does not appear to be justifiable when the relation between weight and power saving achieved and the complexity of modification is considered. The only possible justification would be a critical need for a 25% increase of high resolution area coverage capability, which would be achieved at the expense of multiple stereo capability. It should be noted that the multiple stereo capability may, with addition of filters, be utilized in colorimetry experiments.

Case IV, requiring the removal of the photographic subsystem, results in the following weight, volume and power savings:

Volume	- 5.9 ft.^{3}
Weight	- 149 lbs.
Power	- 80.1 watts during daylight operation
	- 15 watts during nighttime operation

Flight programmer functions used for control of the photo subsystem would be available for control of other experiments.

3.2.7 REMOVAL OF BLOCK I EXPERIMENTS

The deletion of the Block I micrometeoroid detectors and radiation sensors results in weight saving as shown below:

MICROMETEOROID DETECTORS, WIRING AND STRUCTURE

1. Detectors: 20 at .156# each GFE MD-1		3.20#
2. Wiring		.60#
SH 3355 MT 701 thru MT 720 (20 wires) #24 at .00023#/in. 2154"		
SH 3891 SP 891 thru SP 895 (25 wires) #24 at .00023#/in. 338"	.573	
4 Splices YSV-14 at .00269	.011	
2 Clamps at .01	.02	

3.2.7 (Continued)

3. Structure 2.40#

1-06-51 AC 29-41064-009 Clip	(12)	.08		
-011 Filler	(2)	.01		
25-51304-901 Fast		.05		
25-51304-002				
29-41073 Brkt	(12)	.36		
29-41074-001	(14)	.07		
-002	(8)	.04		
-003	(2)	.04		
-004, -005	(60)	0		
25-51830 Angle, Cap	(21)	1.65		
25-51631-900 Fast		.10		
		TOTAL		6.20#

RADIATION DETECTION COMPONENTS, WIRING AND STRUCTURE

1. Two Scintillation Counters MT726, 727 1.37#

   10-72003-2 & 3

2. Logic Box A725 10-72003-4 1.15#

3. Wiring .47#

   SH 3755 #24, #22, #20 wire .294#
   Plug P755 .171#
   4 Splices PSM 18 - P1 at .00073 .003#

   .468

4. Structure .57#

   Pedestals for Scint. Ctr. .23 + .24 .47#
   25-51785-5, -6
   Installation NBR for 2 Ctr. and Logic .10#

TOTAL 3.56#

A power saving of 16 watts is accrued by these deletions.

4.0 EXPERIMENT CONFIGURATIONS AND MISSION PROFILES

This section deals with a limited sample of spacecraft-experiment configurations and mission profiles under ground rules derived from the parametric considerations of the preceding section 3.0 and with consideration of experiment compatibility.

The objective of the following specific case analysis is, in addition to illustration of the applicability of the analysis of the previous section, to provide a more detailed analysis of interrelation between experiments, missions and spacecraft subsystem requirements. These relationships cannot be fully evaluated without establishing a base-line configuration, mission profile and event sequence as will become apparent in the subsequent subsections.

4.1.0 GROUND RULES FOR MECHANICAL LAYOUT AND MISSIONS

Ground rules for mechanical layout and mission operational conduct were established on the basis of minimum spacecraft modification and payload maximization approach. The minimum spacecraft modification approach was primarily dictated by the constraints of the L-5382 Statement of Work relative to total spacecraft weight (860 lbs.-920 lbs.) which precludes major modifications. Secondarily, the minimum modification approach appears to be attractive from the viewpoint of schedule and cost effectiveness. The general ground rules, corresponding to this approach, and their justification are enumerated below:

1. Elliptical lunar orbits, long translunar transit times (90 hrs.) and limited launch periods (3 days/month) will be used in the interest of propellant weight conservation. Off-loaded propellant weight will be used for additional experiment payload.

4.1.0 (Continued)

2. Surface oriented experiments will be rigidly mounted within the spacecraft with their sensor axes parallel to the camera axis. This assumption was made for purposes of the study only and does not represent an actual limitation of mechanical layout except for the obvious field of view obstructions of the tank deck, equipment mounting deck, and the solar panels.

3. Surface directed experiment orientation into the orbital plan and to local vertical will be accomplished, if necessary, by spacecraft maneuvers in a manner analogous to the photographic maneuver. An exception may arise if continuing experiments must be performed over a long length of arc in which case the implementation of a capability for continuous torquing to local vertical (over limited arc length) will be considered in the interest of performance improvement and attitude control gas conservation.

4. Lunar environment experiments will be mounted rigidly to the spacecraft and will be operated continuously, subject to power limitations and interruptions for data transmissions, at the attitude appropriate to the spacecraft operating modes. (Sun-Canopus reference during cruise and local vertical during surface experiment operation.)

5. The useable volume inside of the photographic subsystem made available by deletions specified under Cases II and III of the Statement of Work will not be utilized in order to avoid redesign of the package with each experiment modification and interchange.

4.1.0 (Continued)

6. Data storage for the scientific experiment will be provided with sufficient capacity to utilize effectively the high data rate of the video transmission system, on an intermittent time share basis, in order to minimize communication subsystem modifications.

7. Structural changes will be limited to these required to support additional experiments except for the replacement of the arch supporting structure necessary for camera package removal, by a truss structure in the case of deletion of the photographic subsystem (Case IV). Micrometeoroid detectors and radiation sensors and their supporting structures and cabling will be removed in all cases.

8. The photographic subsystem constraint of 50° - 75° solar illumination at the subspacecraft point will be preserved for Cases I through III of the Statement of Work.

9. Experiment definition of the Statement of Work as reproduced in Appendices A and B will be used in spacecraft configuration studies and will be complemented by data of Appendix C only to the extent necessary for completeness of definition.

With the exception of the changes enumerated above and the addition of programmer command telemetry functions, when specified, all subsystems will remain, inasmuch as possible, unchanged.

4.2.0 SAMPLE EXPERIMENT GROUPINGS

Sample experiment groupings, for each of the Cases I through IV

4.2.0 (Continued)

specified by the L-5382 Statement of Work, were defined on the basis of capabilities indicated by the parametric studies, experiment descriptions of Appendices A and B and following consultation with the contracting agency.

These experiments groupings, their total weight and structural integration allowances based on weight availability estimates, are shown in the tabulation of Figure 4.2.0.1. The corresponding weight availability estimates and associated approximate orbital profiles are summarized in Figure 4.2.0.2.

4.3.0 MECHANICAL LAYOUTS, WEIGHTS AND CENTER OF GRAVITY

Configuration drawings were generated for five of the cases. These configurations are shown as isometric drawings of experiment arrangements in two views in Figures 4.3.0.1 through 4.3.0.10, respectively. The corresponding cross-sectional views are shown in Figures 4.3.0.11 through 4.3.0.21. (Pages 246 through 256 inclusive)

On the basis of the above layouts, corrected weight statements were obtained for all ten experiments. These are tabulated in Figures 4.3.0.22 through 4.3.0.31, and include moments of inertia and c.g. data in the critical cases.

Since the weight statements substantiated the initial estimates of feasibility of mechanical integration of the experiments, in conjunction with the above five configuration layouts, it appeared unnecessary to generate additional configuration data.

EXPERIMENT GROUPING WEIGHT

CASE	I			II			III			IV
EXPERIMENT	A	B	C	A	B	C	B	A	C	
Micrometeoroid		27	-	-	-	-	-	-	-	27
Solar Plasma	12		12	-	-	-	-	-	-	12
Magnetometer	12	12	12	-	-	-	-	-	-	12
Photometry/Colorimetry	4	4	-	4	4	4	4	4	-	4
Gamma Ray	-	-	28	-	28	-	-	-	28	28
Radiometry	-	-	-	6	-	6	-	6	-	6
X-Ray	-	-	-	18	-	18	-	-	-	18
I.R.	-	-	-	-	4	4	-	-	-	4
Selenodesy	-	-	-	-	-	-	-	-	-	-
Bi-Static Radar	-	-	-	-	-	-	5	5	-	5
Total Experiments	28	55	52	28	36	32	9	15	28	116
Structures and Cabling Allowance	24	35	38	7	19	23	4	8	10	69
Payload Weight Estimated	52	90	90	35	55	55	13	23	38	185

Fig. 4.2.0.1

AVAILABILITY OF WEIGHT FOR EXPERIMENTS

CASE	I A	I B	I C	II A	II B	II C	III B	III A	III C	IV A	IV B	IV C
Deletions												
Micrometeoroid and Radiation Detectors	10	10	10	10	10	10	10	10	10	10	10	10
Hi-Res. Photo Cap.	-	-	-	25	25	25	-	-	-	-	-	-
Lo-Res. Photo Cap.	-	-	-	-	-	-	3	3	3	-	-	-
Total Photo	-	-	-	-	-	-	-	-	-	149	149	149
Additional Boost Cap.	60	60	60	-	-	-	-	-	-	-	-	-
Sub-Total	70	70	70	35	35	35	13	13	13	159	159	159
Fuel Off-Loading *	-	20	20	-	20	20	-	10	25	-	-	-
Battery Retrofit (20 amp. hr.)	-20	-	-	-	-	-	-	-	-	-	-	-
TOTAL	50	90	90	35	55	55	13	23	38	185	185	185
* **Mission Profile Variation**												
Perilune Altitude (approx.)	46	184	184	46	184	184	46	46	184			
Apolune Altitude (approx.)	2300	3000	3000	1850	3000	3000	1850	3500	3000	3000		
Inclination (approx.)	10°	35°	35°	33°	45°	45°	15°	10°	45°	45°		

Fig. 4.2.0.2

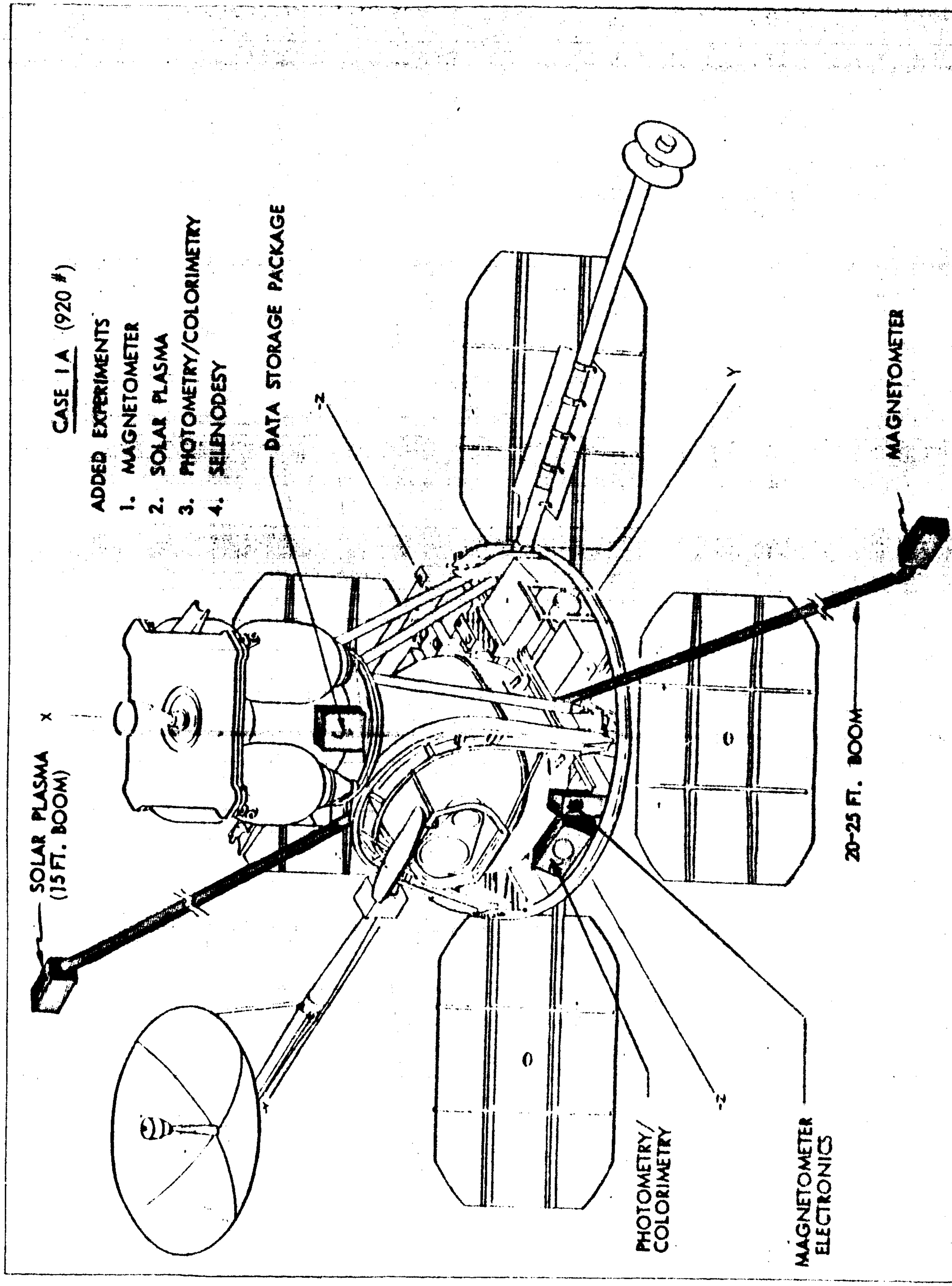

CASE I A (920 #)
ADDED EXPERIMENTS
1. MAGNETOMETER
2. SOLAR PLASMA
3. PHOTOMETRY/COLORIMETRY
4. SELENODESY
DATA STORAGE PACKAGE
-Z
Y
X
SOLAR PLASMA
(15 FT. BOOM)
MAGNETOMETER
20-25 FT. BOOM
PHOTOMETRY/
COLORIMETRY
Z
MAGNETOMETER
ELECTRONICS

Fig.4.3.0.1

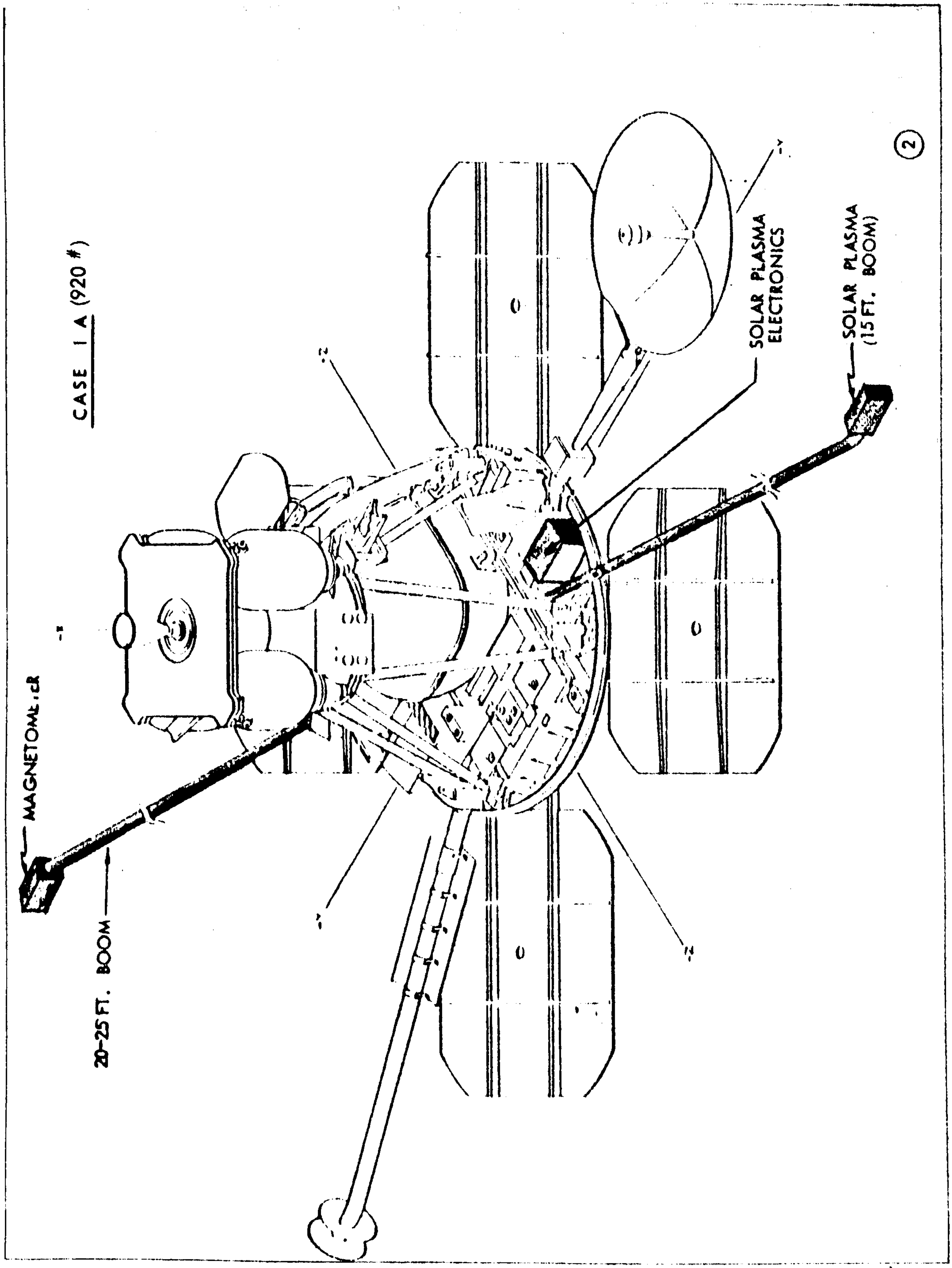

CASE 1 A (920 #)
MAGNETOMETER
20-25 FT. BOOM
SOLAR PLASMA ELECTRONICS
SOLAR PLASMA (15 FT. BOOM)
2

Fig. 4.3.0.2

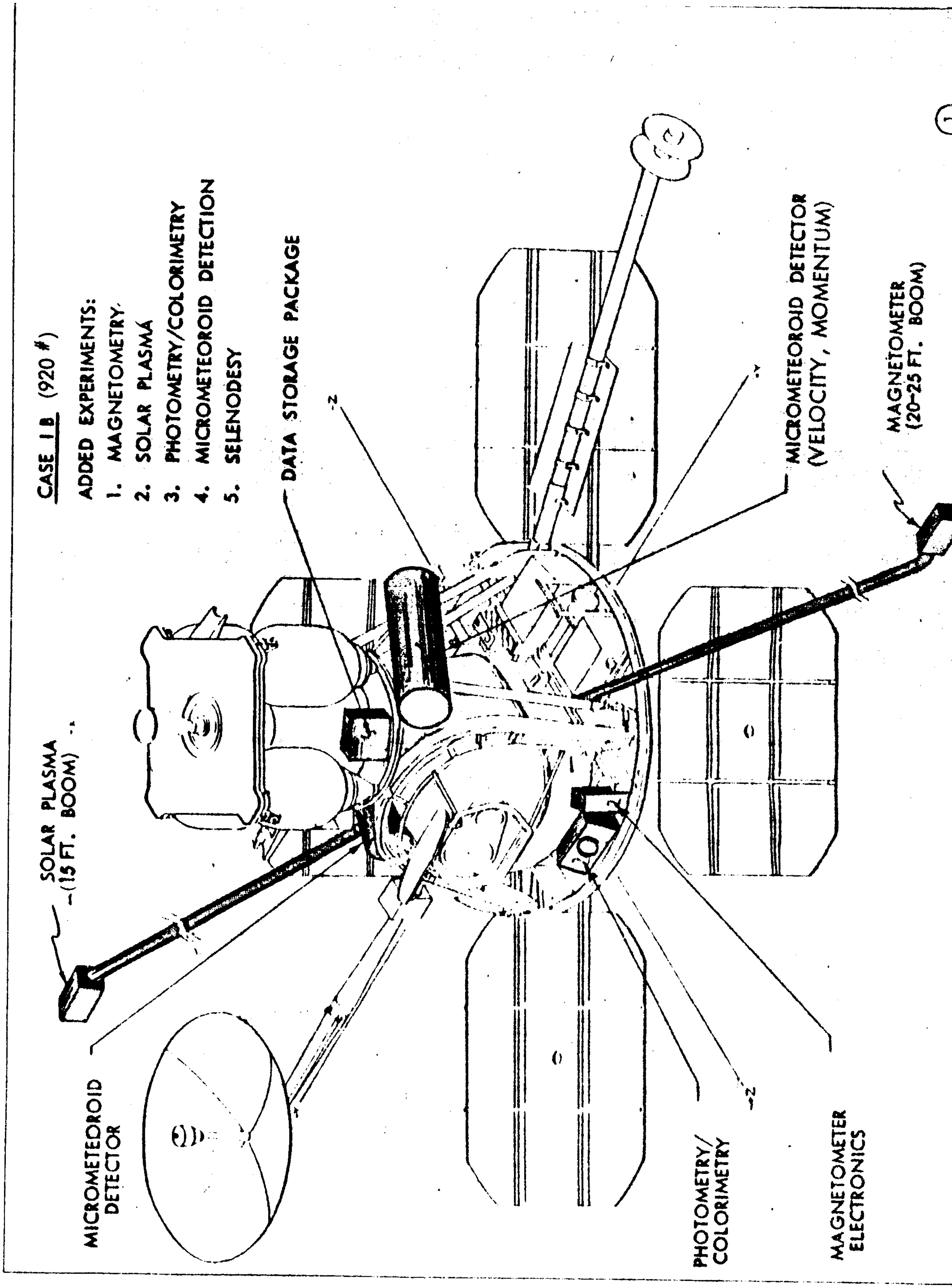

CASE I B (920 #)
ADDED EXPERIMENTS:
1. MAGNETOMETRY
2. SOLAR PLASMA
3. PHOTOMETRY/COLORIMETRY
4. MICROMETEOROID DETECTION
5. SELENODESY
DATA STORAGE PACKAGE
MICROMETEOROID DETECTOR (VELOCITY, MOMENTUM)
MAGNETOMETER (20-25 FT. BOOM)
SOLAR PLASMA (15 FT. BOOM)
MICROMETEOROID DETECTOR
PHOTOMETRY/ COLORIMETRY
MAGNETOMETER ELECTRONICS
-Z
Y
-Z
1

Fig. 4.3.0.3

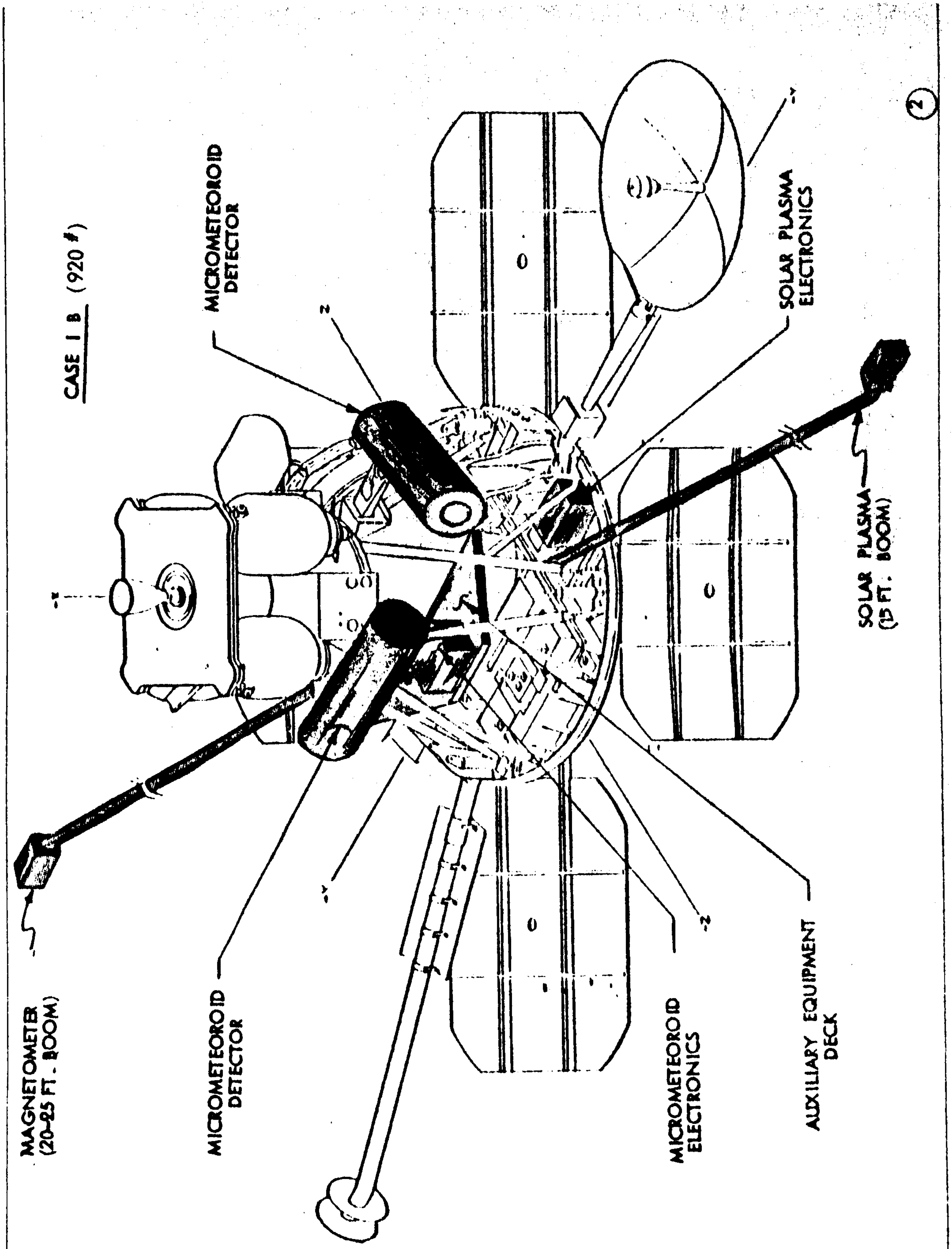

CASE I B (920 #)
MICROMETEOROID DETECTOR
SOLAR PLASMA ELECTRONICS
SOLAR PLASMA (15 FT. BOOM)
MAGNETOMETER (20-25 FT. BOOM)
MICROMETEOROID DETECTOR
MICROMETEOROID ELECTRONICS
AUXILIARY EQUIPMENT DECK
2

Fig. 4.3.0.4

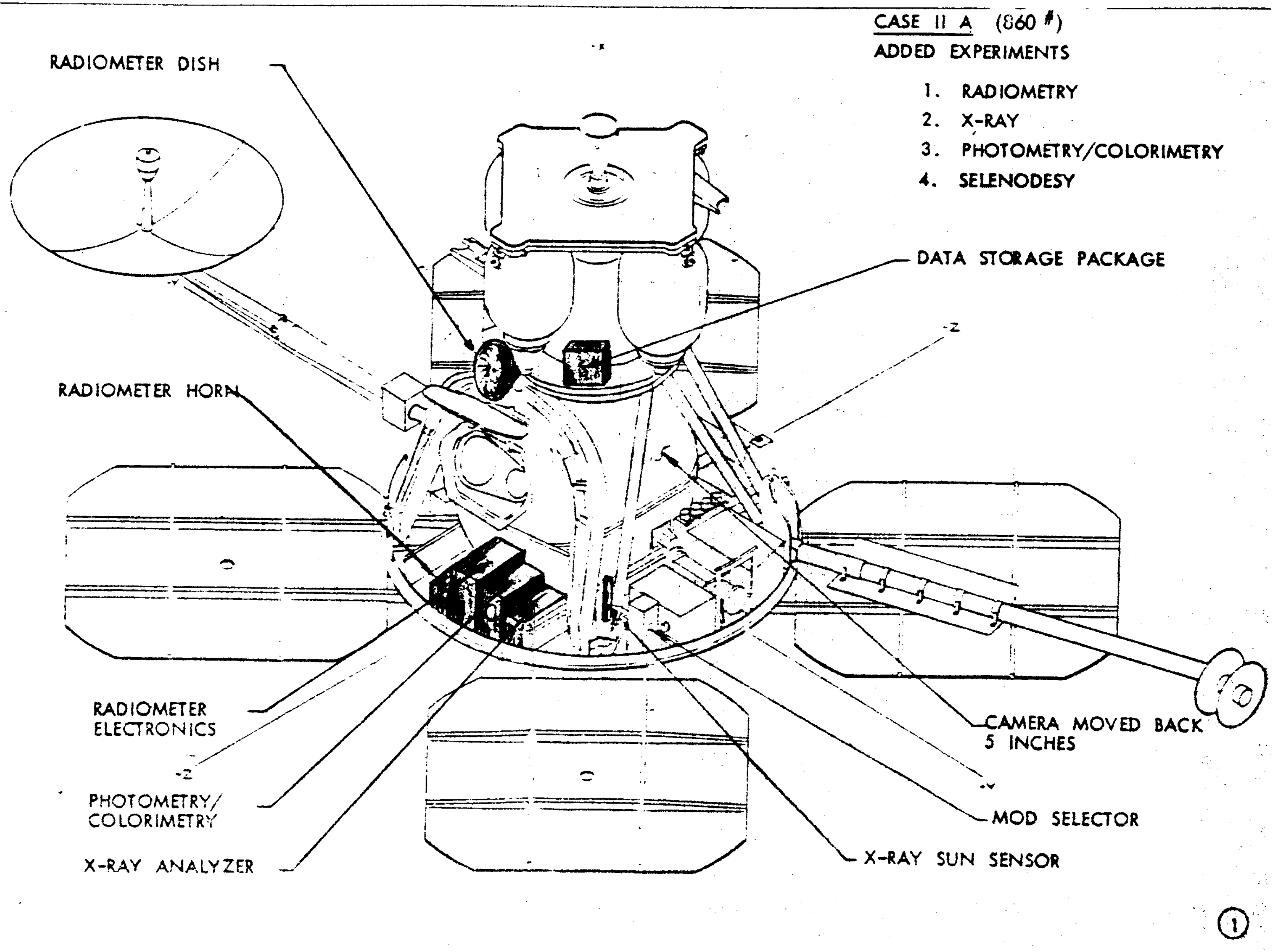

CASE II A (860 #)
ADDED EXPERIMENTS
1. RADIOMETRY
2. X-RAY
3. PHOTOMETRY/COLORIMETRY
4. SELENODESY
RADIOMETER DISH
DATA STORAGE PACKAGE
-Z
RADIOMETER HORN
RADIOMETER ELECTRONICS
-Z
PHOTOMETRY/ COLORIMETRY
X-RAY ANALYZER
CAMERA MOVED BACK 5 INCHES
-Y
MOD SELECTOR
X-RAY SUN SENSOR
1

Fig. 4.3.0.5

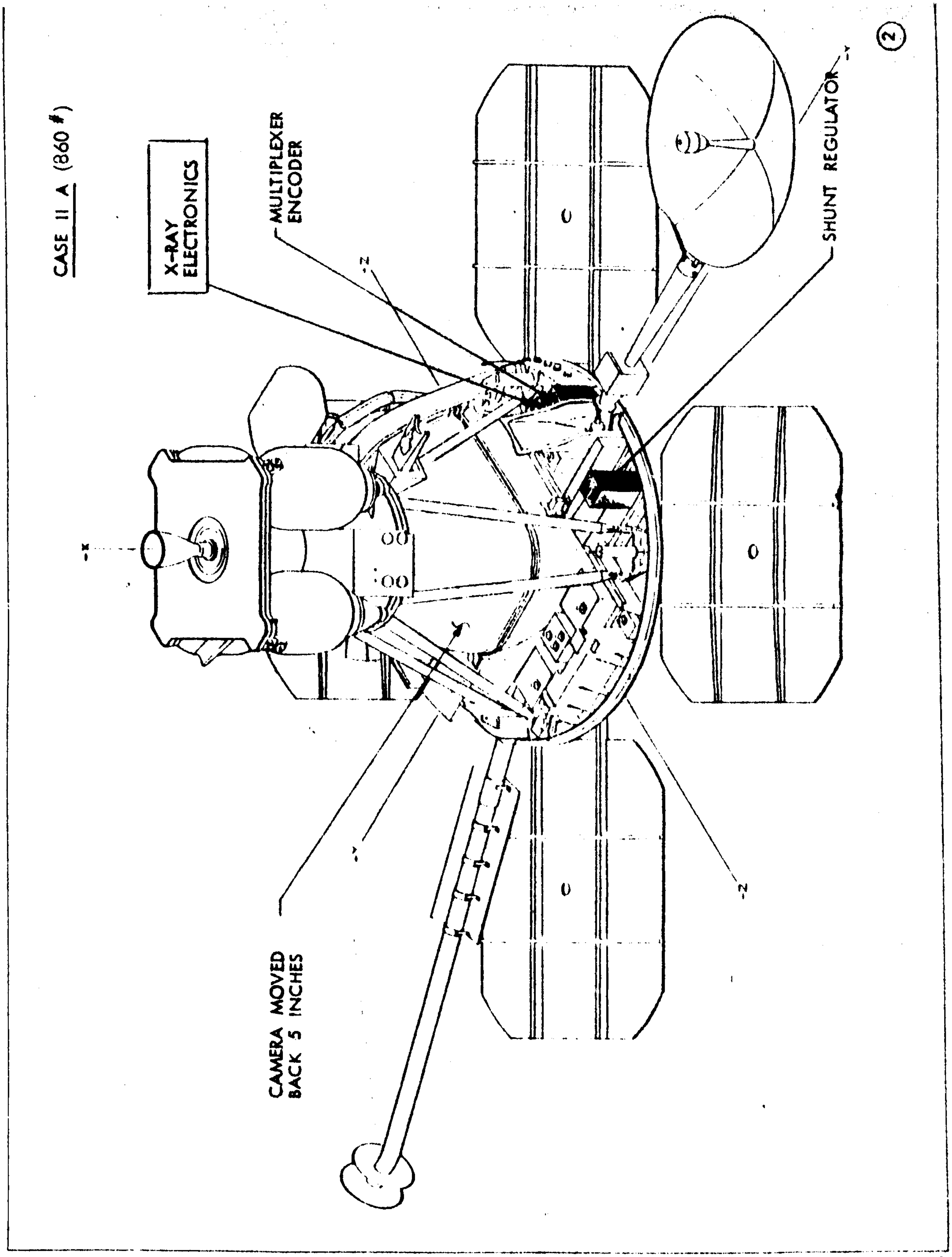
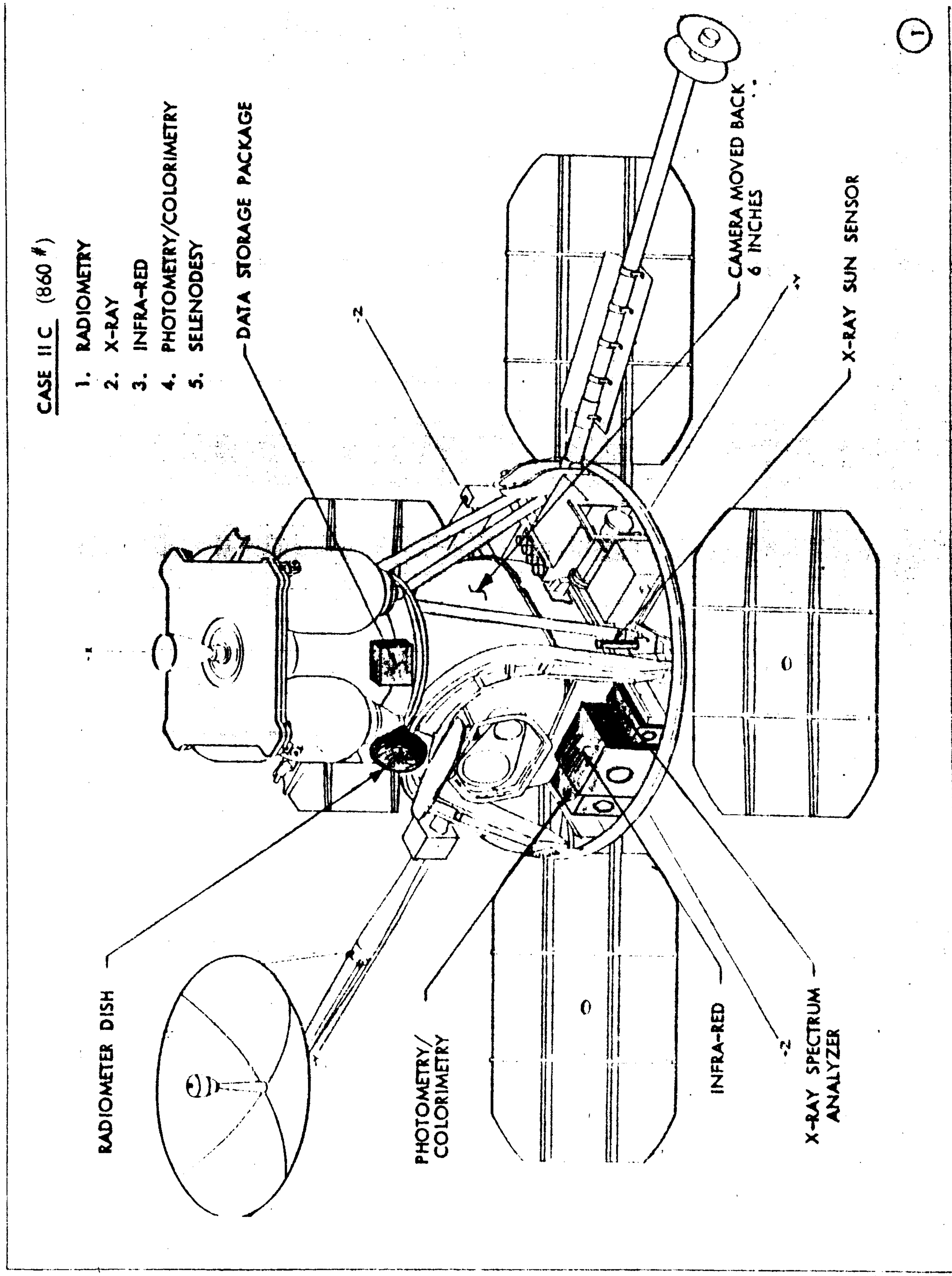

CASE II A (860 #)
X-RAY ELECTRONICS
MULTIPLEXER ENCODER
SHUNT REGULATOR
CAMERA MOVED BACK 5 INCHES
-X
-Z
-Y
Y
Z
2

Fig. 4.3.0.6



1
CASE II C (860 #)
1. RADIOMETRY
2. X-RAY
3. INFRA-RED
4. PHOTOMETRY/COLORIMETRY
5. SELENODESY
DATA STORAGE PACKAGE
CAMERA MOVED BACK 6 INCHES
X-RAY SUN SENSOR
RADIOMETER DISH
PHOTOMETRY/ COLORIMETRY
INFRA-RED
X-RAY SPECTRUM ANALYZER

Fig. 4.3.0.7

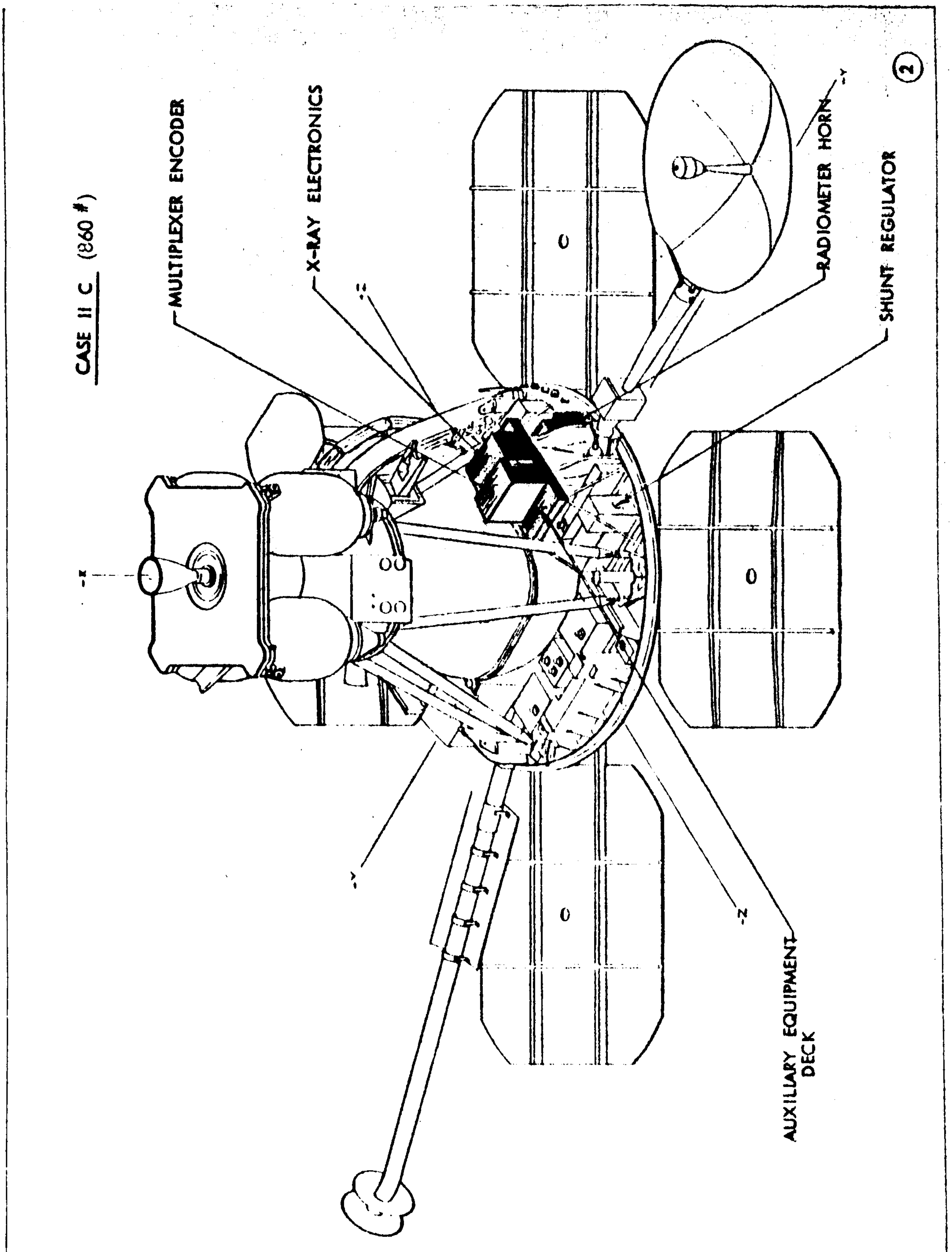

CASE II C (860 #)
MULTIPLEXER ENCODER
X-RAY ELECTRONICS
RADIOMETER HORN
SHUNT REGULATOR
AUXILIARY EQUIPMENT DECK
2

Fig. 4.3.0.8

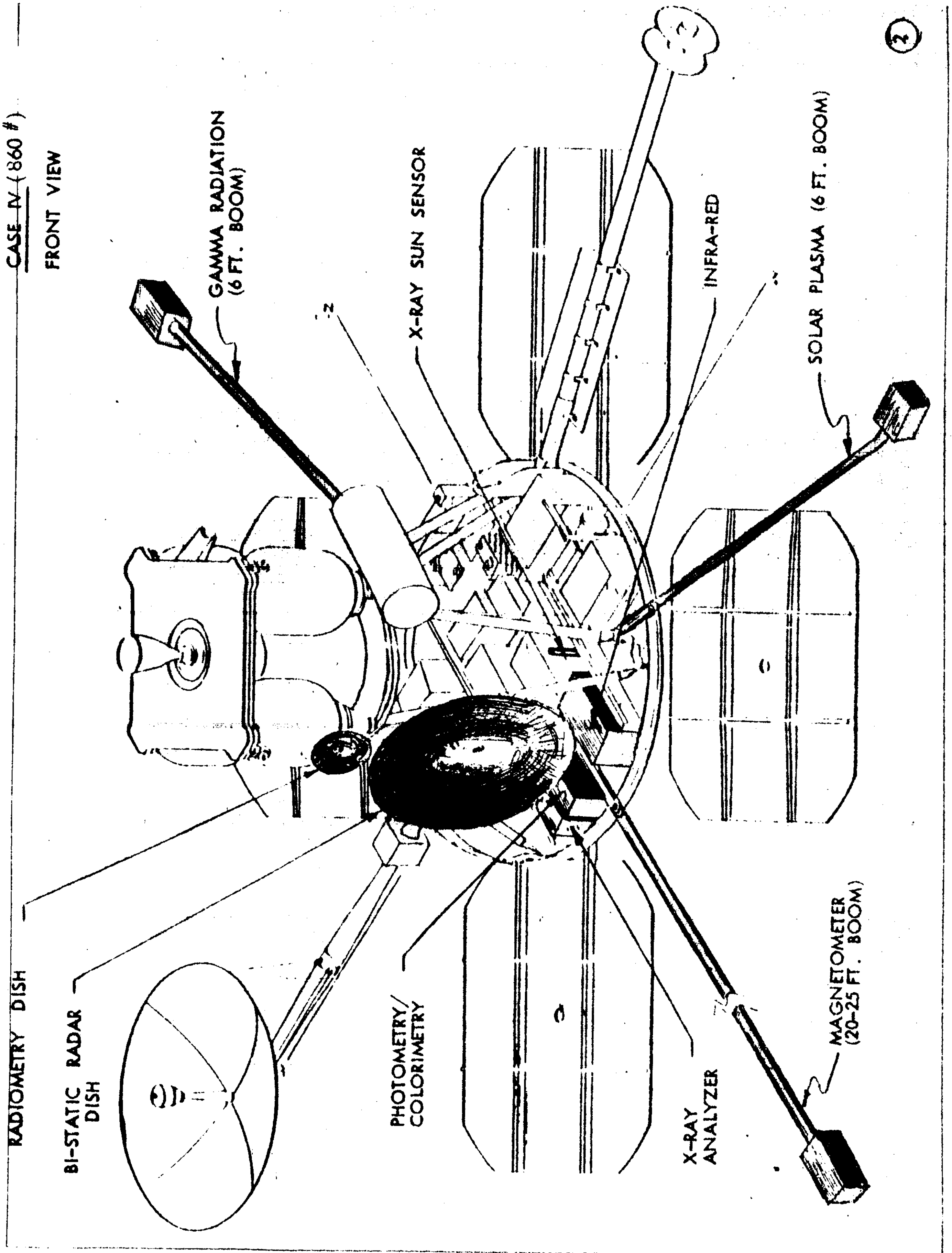
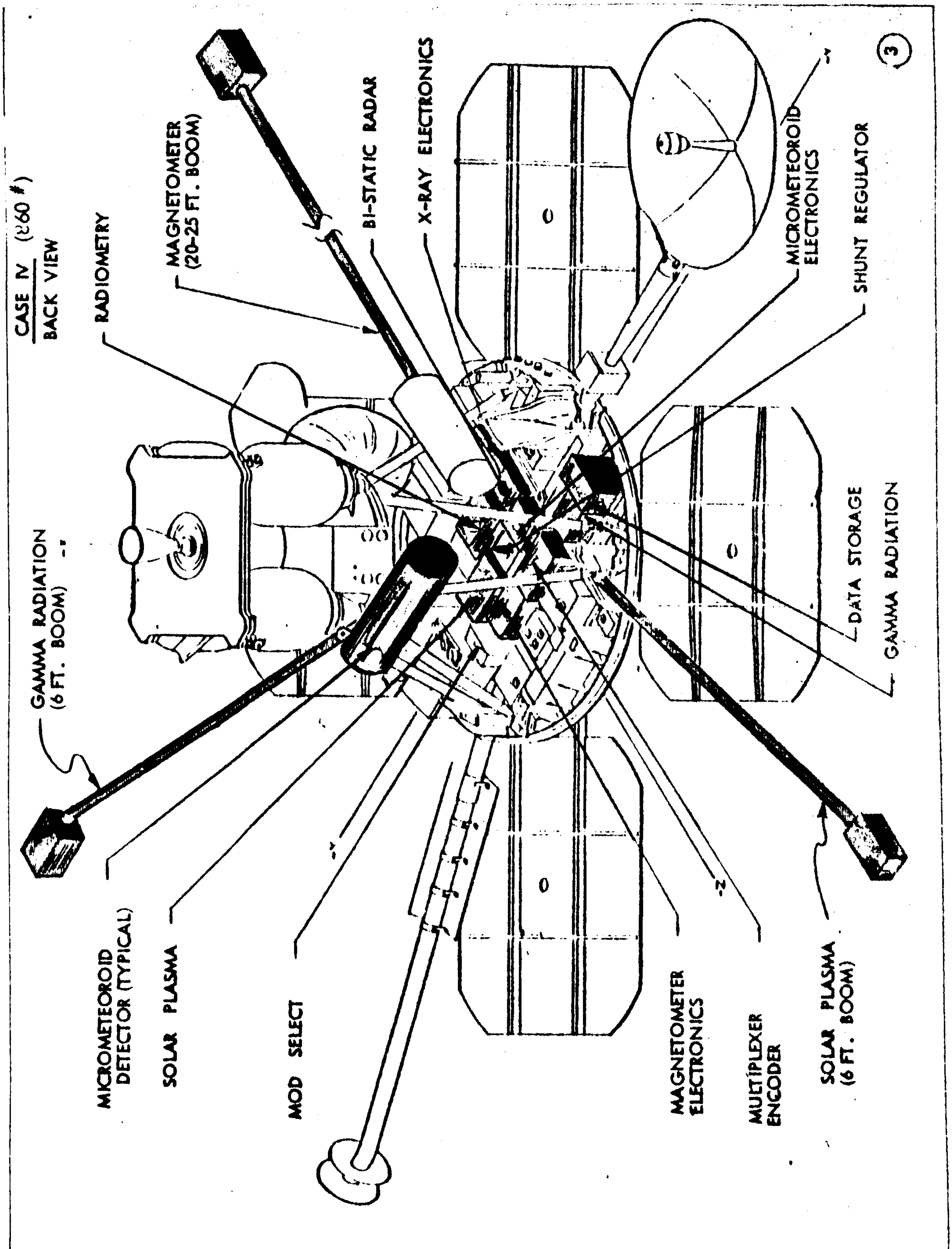

CASE IV (860 #)
FRONT VIEW
GAMMA RADIATION (6 FT. BOOM)
X-RAY SUN SENSOR
INFRA-RED
SOLAR PLASMA (6 FT. BOOM)
RADIOMETRY DISH
BI-STATIC RADAR DISH
PHOTOMETRY/ COLORIMETRY
X-RAY ANALYZER
MAGNETOMETER (20-25 FT. BOOM)
2

Fig. 4.3.0.9



CASE IV (≈60 #)
BACK VIEW
RADIOMETRY
MAGNETOMETER (20-25 FT. BOOM)
BI-STATIC RADAR
X-RAY ELECTRONICS
MICROMETEOROID ELECTRONICS
SHUNT REGULATOR
3
GAMMA RADIATION (6 FT. BOOM)
DATA STORAGE
GAMMA RADIATION
MICROMETEOROID DETECTOR (TYPICAL)
SOLAR PLASMA
MOD SELECT
MAGNETOMETER ELECTRONICS
MULTIPLEXER ENCODER
SOLAR PLASMA (6 FT. BOOM)

Fig. 4.3.0.10

CASE Ia

	WEIGHT -- LB EQUIPMENT & SENSORS	STRUCTURE & WIRING	TOTALS
7/1/65 L/O INERT WEIGHT			572.08
DELETE			-39.71
Radiation Detection	2.52	1.03	
Micrometeoroid Detection	3.20	3.00	
12 Amp-Hr Batteries	29.96		
ADD			116.65
20 Amp-Hr Batteries	51.00	2.00	
Wire For Re-Arranged Boxes		.33	
Photometer - Colorimeter	4.00	.90	
Magnetometer	12.00	12.26	
Solar Plasma	12.00	12.16	
Data Storage Tape Recorder	9.00	1.00	
TOTAL INERT WEIGHT			649.02
ADD			273.05
N_2 -- 11 months	4.29		
N_2 -- 30 days	6.29		
Propellant	262.47		
TOTAL WEIGHT @ AGENA SEPARATION			922.07
TOTAL WEIGHT ALLOWED PER NASA RFP			920.00

CASE Ia @ (Solar Panels & Booms Deployed)	WEIGHT LB	CENTER OF GRAVITY IN X	Z	Y	INERTIA SLUG - FT^2 I_{y-y}	I_{x-x}	I_{z-z}
INERT WEIGHT	649.02	222.66	-.12	+.02	214.27	292.14	168.28
AGENA SEPARATION	922.07	212.75	-.08	+.01	260.58	301.75	217.78

Fig. 4.3.0.22

CASE Ib

	WEIGHT -- LB EQUIPMENT & SENSORS	STRUCTURE & WIRING	TOTALS
7/1/65 L/O INERT WEIGHT			572.08
DELETE			- 9.75
Radiation Detection	2.52	1.03	
Micrometeoroid Detection	3.20	3.00	
ADD			97.65
Photometer - Colorimeter	4.00	.9	
Magnetometer	12.00	12.26	
Solar Plasma	12.00	12.16	
Data Storage Tape Recorder	9.00	1.00	
Wire For Re-Arranged Boxes		.33	
Micrometeoroid Detector	27.00	3.20	
Auxiliary Equipment Deck		3.80	
TOTAL INERT WEIGHT			659.98
ADD			253.05
N_2 -- 11 months	4.29		
N_2 -- 30 days	6.29		
Propellant	242.47		
TOTAL WEIGHT @ AGENA SEPARATION			913.03
TOTAL WEIGHT ALLOWED PER NASA RFP			920.00

Fig. 4.3.0.23

CASE Ic

	WEIGHT -- LB EQUIPMENT & SENSORS	STRUCTURE & WIRING	TOTALS
7/1/65 L/O INERT WEIGHT			572.08
DELETE			- 9.75
Radiation Detection	2.52	1.03	
Micrometeoroid Detection	3.20	3.00	
ADD			92.08
Solar Plasma	12.00	8.61	
Magnetometer	12.00	12.26	
Gamma Ray	28.00	9.21	
Data Storage Tape Recorder	9.00	1.00	
TOTAL INERT WEIGHT			654.41
ADD			273.05
N_2 -- 11 months	4.29		
N_2 -- 30 days	6.29		
Propellant	262.47		
TOTAL WEIGHT @ AGENA SEPARATION			917.46
TOTAL WEIGHT ALLOWED PER NASA RFP			920.00

Fig. 4.3.0.24

CASE IIa

	WEIGHT -- LB EQUIPMENT & SENSORS	STRUCTURE & WIRING	TOTALS
7/1/65 L/O INERT WEIGHT			572.08
DELETE			-37.75
Radiation Detection	2.52	1.03	
Micrometeoroid Detection	3.20	3.00	
Camera High Resolution	28.00		
ADD			46.18
Wire For Re-Arranged Boxes		.98	
Auxiliary Equipment Deck		2.30	
Radiometer	6.00	2.10	
X -Ray	18.00	1.90	
Photometer - Colorimeter	4.00	.90	
Data Storage Tape Recorder	9.00	1.00	
TOTAL INERT WEIGHT			580.51
ADD			273.05
N_2 -- 11 months	4.29		
N_2 -- 30 days	6.29		
Propellant	262.47		
TOTAL WEIGHT @ AGENA SEPARATION			853.56
TOTAL WEIGHT ALLOWED PER NASA RFP			860.00

Fig. 4.3.0.25

CASE IIb

	WEIGHT -- LB EQUIPMENT & SENSORS	STRUCTURE & WIRING	TOTALS
7/1/65 L/O INERT WEIGHT			572.08
DELETE			-37.75
Radiation Detection	2.52	1.03	
Micrometeoroid Detection	3.20	3.00	
Camera High Resolution	28.00		
ADD			57.99
Wire For Re-Arranged Boxes		.98	
Gamma Ray	28.00	9.21	
Photometer - Colorimeter	4.00	.90	
Infra-Red	4.00	.90	
Data Storage and Tape Recorder	9.00	1.00	
TOTAL INERT WEIGHT			592.32
ADD			253.05
N_2 -- 11 months	4.29		
N_2 -- 30 days	6.29		
Propellant	242.47		
TOTAL WEIGHT @ AGENA SEPARATION			845.37
TOTAL WEIGHT ALLOWED PER NASA RFP			860.00

Fig. 4.3.0.26

USE FOR TYPEWRITTEN MATERIAL ONLY

CASE IIc

	WEIGHT -- LB EQUIPMENT & SENSORS	STRUCTURE & WIRING	TOTALS
7/1/65 L/O INERT WEIGHT			572.08
DELETE			-37.75
Radiation Detection	2.52	1.03	
Micrometeoroid Detection	3.20	3.00	
Camera High Resolution	28.00		
ADD			52.13
Wire For Re-Arranged Boxes		.98	
Auxiliary Equipment Deck		3.35	
Radiometer	6.00	2.10	
X-Ray	18.00	1.90	
Photometer - Colorimeter	4.00	.90	
Infra-Red	4.00	.90	
Data Storage Tape Recorder	9.00	1.00	
TOTAL INERT WEIGHT			586.46
ADD			253.05
N_2 -- 11 months	4.29		
N_2 -- 30 days	6.29		
Propellant	242.47		
TOTAL WEIGHT @ AGENA SEPARATION			839.51
TOTAL WEIGHT ALLOWED PER NASA RFP			860.00

Fig. 4.3.0.27

USE FOR TYPEWRITTEN MATERIAL ONLY

CASE IIIa

	WEIGHT -- LB EQUIPMENT & SENSORS	WEIGHT -- LB STRUCTURE & WIRING	TOTALS
7/1/65 L/O INERT WEIGHT			572.08
DELETE			-12.75
Radiation Detection	2.52	1.03	
Micrometeoroid Detection	3.20	3.00	
Camera Low Resolution	3.00		
ADD			38.83
Wire For Re-Arranged Boxes		.50	
Photometer - Colorimeter	4.00	.90	
Bi-Static Radar	5.00	10.33 (Incl. 30" Antenna)	
Radiometer	6.00	2.10	
Data Storage Tape Recorder	9.00	1.00	
TOTAL INERT WEIGHT			598.16
ADD			263.05
N_2 -- 11 months	4.29		
N_2 -- 30 days	6.29		
Propellant	252.47		
TOTAL WEIGHT @ AGENA SEPARATION			861.21
TOTAL WEIGHT ALLOWED PER NASA RFP			860.00

Fig. 4.3.0.28

CASE IIIb

	WEIGHT -- LB EQUIPMENT & SENSORS	STRUCTURE & WIRING	TOTALS
7/1/65 L/O INERT WEIGHT			572.08
DELETE			-12.75
Radiation Detection	2.52	1.03	
Micrometeoroid Detection	3.20	3.00	
Camera Low Resolution	3.00		
ADD			30.73
Wire For Re-Arranged Boxes		.50	
Photometer - Colorimeter	4.00	.90	
Bi-Static Radar	5.00	10.33 (Incl. 30" Antenna)	
Data Storage Tape Recorder	9.00	1.00	
TOTAL INERT WEIGHT			590.06
ADD			263.05
N_2 -- 11 months	4.29		
N_2 -- 30 days	6.29		
Propellant	252.47		
TOTAL WEIGHT @ AGENA SEPARATION			853.11
TOTAL WEIGHT ALLOWED PER NASA RFP			860.00

Fig. 4.3.0.29

CASE IIIc

	WEIGHT -- LB EQUIPMENT & SENSORS	STRUCTURE & WIRING	TOTALS
7/1/65 L/O INERT WEIGHT			572.08
DELETE			-12.75
Radiation Detection	2.52	1.03	
Micrometeoroid Detection	3.20	3.00	
Camera Low Resolution	3.00		
ADD			47.21
Gamma Ray	28.00	9.21	
Data Storage Tape Recorder	9.00	1.00	
TOTAL INERT WEIGHT			606.54
ADD			248.05
N_2 -- 11 months	4.29		
N_2 -- 30 days	6.29		
Propellant	237.47		
TOTAL WEIGHT @ AGENA SEPARATION			854.59
TOTAL WEIGHT ALLOWED PER NASA RFP			860.00

Fig. 4.3.0.30

USE FOR TYPEWRITTEN MATERIAL ONLY

CASE IV

	WEIGHT -- LB EQUIPMENT & SENSORS	STRUCTURE & WIRING	TOTALS
7/1/65 L/O INERT WEIGHT			572.08
DELETE			-163.96
Radiation Detection	2.52	1.03	
Micrometeoroid Detection	3.20	3.00	
Photo Subsystem	145.54	4.88	
Structural Arch		3.79	
ADD			176.88
Truss Tubes		1.04	
Photometer - Colorimeter	4.00	.90	
Infra-Red	4.00	.90	
Radiometer	6.00	2.10	
X-Ray	18.00	2.00	
Micrometeoroid Detector	27.00	3.20	
Bi-Static Radar	5.00	10.33 (Incl. 30" Antenna)	
Solar Plasma	12.00	8.61	
Gamma Ray	28.00	9.21	
Magnetometer	12.00	12.26	
Data Storage Tape Recorder	9.00	1.00	
Wire For Re-Arranged Boxes		.33	
TOTAL INERT WEIGHT			585.00
ADD			273.05
N_2 -- 11 months	4.29		
N_2 -- 30 days	6.29		
Propellant	262.47		
TOTAL WEIGHT @ AGENA SEPARATION			858.05
TOTAL WEIGHT ALLOWED PER NASA RFP			860.00

CASEIV @ (Solar Panels & Booms Deployed)	WEIGHT LB	CENTER OF GRAVITY IN X	Z	Y	INERTIA SLUG - FT^2 I_{y-y}	I_{x-x}	I_{z-z}
INERT WEIGHT	585.00	226.65	+.16	-.13	205.39	224.69	111.23
AGENA SEPARATION	858.05	212.24	+.11	-.09	251.70	234.30	160.73

Fig. 4.3.0.31

4.4.0 MISSION PROFILES

The choise of mission profiles for a detailed analysis was, by necessity, limited to two configurations. This was accomplished using the following rationale:

1. From the viewpoint of weight and volume carrying capability none of the configurations considered appeared to offer a critical need for examination.

2. The "black box" definition of experiments and their low power requirements appeared to indicate no critical problems in the thermal area for any particular experiment.

3. From the viewpoint of the attitude control subsystem the critical configurations appeared to be these configurations requiring boom deployment, resulting in partly changed moments of inertia, and those missions requiring a large number of experiment attitude maneuvers and/or long arc coverages in the course of a single orbital pass. These items will be less critical if the angular accelerations and control tolerances may be relaxed from those of the photo subsystem.

4. From the viewpoint of the power subsystem, continuous operation of experiments during "nighttime",operation of groups of experiments over long arc lengths (departure from cruise attitude) and operation of the high power transmitter for data experiment data transmission in addition to video data, appeared to be the most critical areas.

4.4.0 (Continued)

5. From the viewpoint of the communication subsystem, under the ground rules defined under 4.1.0, the critical problem areas appeared to exist in the cases of high experiment data requirements over short arc lengths, occurring in surface directed experiment, and/or accumulation of low data rates as in the case of environmental experiments operating continuously during the sun and earth occultation times by the moon (relative to the spacecraft).

On the basis of the above considerations the configurations of Cases IB and IIC were chosen to cover the range of critical factors.

The initial conditions for the mission profiles for these configurations were chosen as follows:

	CASE IB	CASE IIC
Apolune Altitude	3000 km	3000 km
Perilune Altitude	92 km	46 km
Inclination	33°	45°
Illumination at Perilune	60°	90°
Photographic Altitude	92 km	184 km
Photographic Resolution	2 meters 16 meter stereo	32 meter stereo
IR Altitude	--	46 km
Radiometry Altitude	--	46 km
Photometry/Colorimetry Altitude	92 km	184 km
X-Ray Altitude	--	50 km
Photography Illumination	50° - 75°	50° - 75°
IR Illumination	--	105° - 90°
Radiometry Illumination	--	80° - 60°
Photometry/Colorimetry Illumination	50° - 75°	50° - 75°
Photo Target Area	Aristarchus	Near Equatorial Band (25° x 360°)
IR Target Area	--	Near 26° Latitude Band (13° x 360°)
Photometry/Colorimetry Target Area	Near 26° L. band (25° x 360°)	Near Equatorial Band (25° x 360°)
Radiometry Target Area	--	Near 26° N Latitude Band (25° x 360°)
X-Ray Target Area	--	Near 22° N Latitude Band (20° x 360°)

4.4.0 (Continued)

The above profiles are illustrated in Figures 4.4.0.1 and 4.4.0.2, respectively. The detailed orbital data, corresponding to these orbital profiles, is shown in Figures 4.4.0.3 through 4.4.0.15 and the corresponding mission event sequences are summarized in Figures 4.4.0.16 and 4.4.0.17, respectively. The area coverages of mission IIC relative to each of the surface oriented experiments are illustrated in Figure 4.4.0.18.

4.4.1 CASE IB

The mission profile and mission event sequence for Case IB is generally similar to the standard Lunar Orbiter mission with the exception of the increased apolune and perilune altitudes and inclination. The difference manifests itself in decreased spacecraft occultation times with respect to both the sun and the earth. The event sequence, prior to the final lunar orbit and initiation of experiments, will correspond closely to the standard Lunar Orbiter mission with the exception of boom deployment sequencing. During the final orbital phase the environmental experiments will be operated continuously, without interfering with the surface related experiments, and the photometry/colorimetry experiment will be performed in conjunction with the photographic experiment (Mapping of Aristarchus) on a sequence of 12 consecutive lunar orbits. The photometry/colorimetry experiment will be additionally performed on each of the final lunar orbits, over the illumination band of 50° - 75°, during the 30 day mission from cruise altitude. This will result in photometry/colorimetry coverage band of an approximately 25° width around the lunar perimeter (360° coverage). Coverage contiguity for the photometry/colorimetry experiment could be

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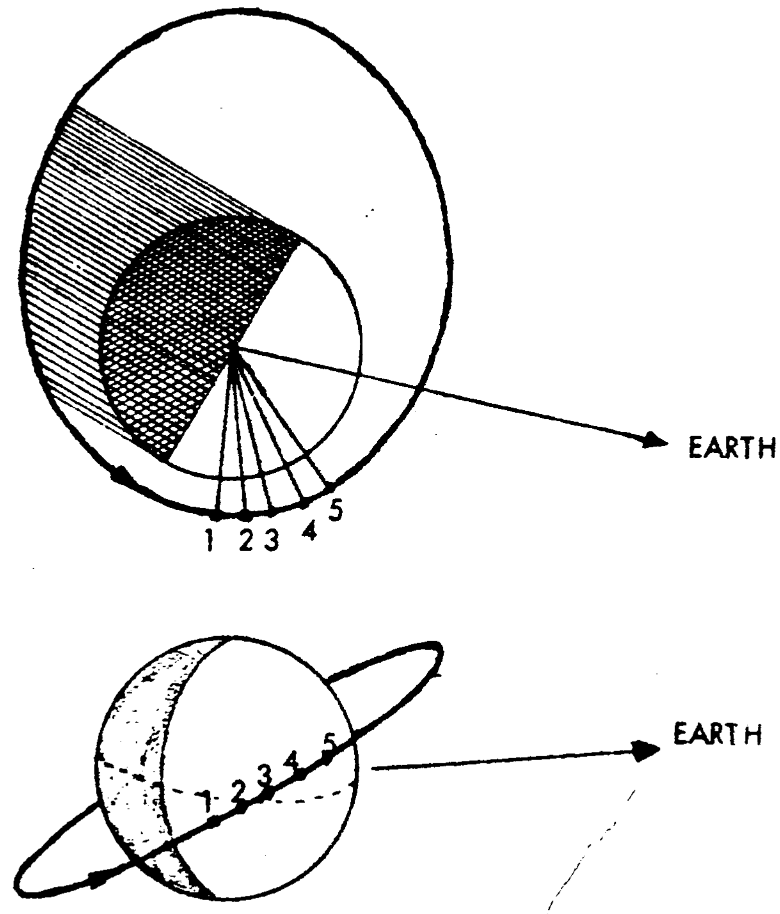
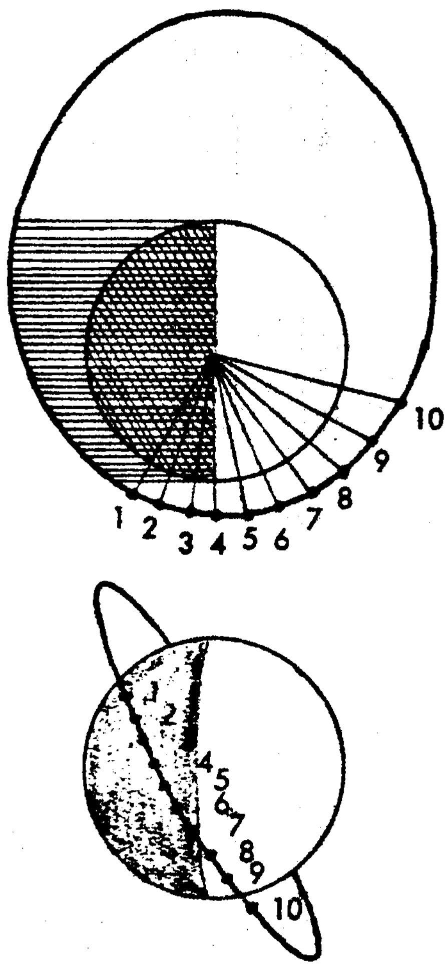
CASE IB MISSION



1 → 2	PHOTO MANEUVER
2 → 3	PHOTOGRAPHY/PHOTOMETRY/ COLORIMETRY
3 → 4	MANEUVER TO CRUISE ATTITUDE
4 → 5	TRANSMIT ENVIRONMENTAL EXPERIMENT DATA
5 → 4	EXPERIMENTS OPERATIONAL
5 + 80 min	PHOTO DATA PROCESSING

Fig. 4.4.0.1

CASE IIC MISSION



1 → 2	MANEUVER TO EXPERIMENT ATTITUDE
2 → 3	IR EXPERIMENT OPERATIONAL
2 → 4	RADIOMETRY EXPERIMENT OPERATIONAL
4 → 6	X-RAY EXPERIMENT OPERATIONAL
5 → 7	PHOTOGRAPHY/PHOTOMETRY/ COLORIMETRY EXPERIMENTS OPERATIONAL
7 → 8	MANEUVER TO CRUISE ATTITUDE
8 → 9	TRANSMIT EXPERIMENT DATA
9 → 10	PROCESS PHOTO DATA
10 →	TRANSMIT :4 FRAMES OF PHOTOS

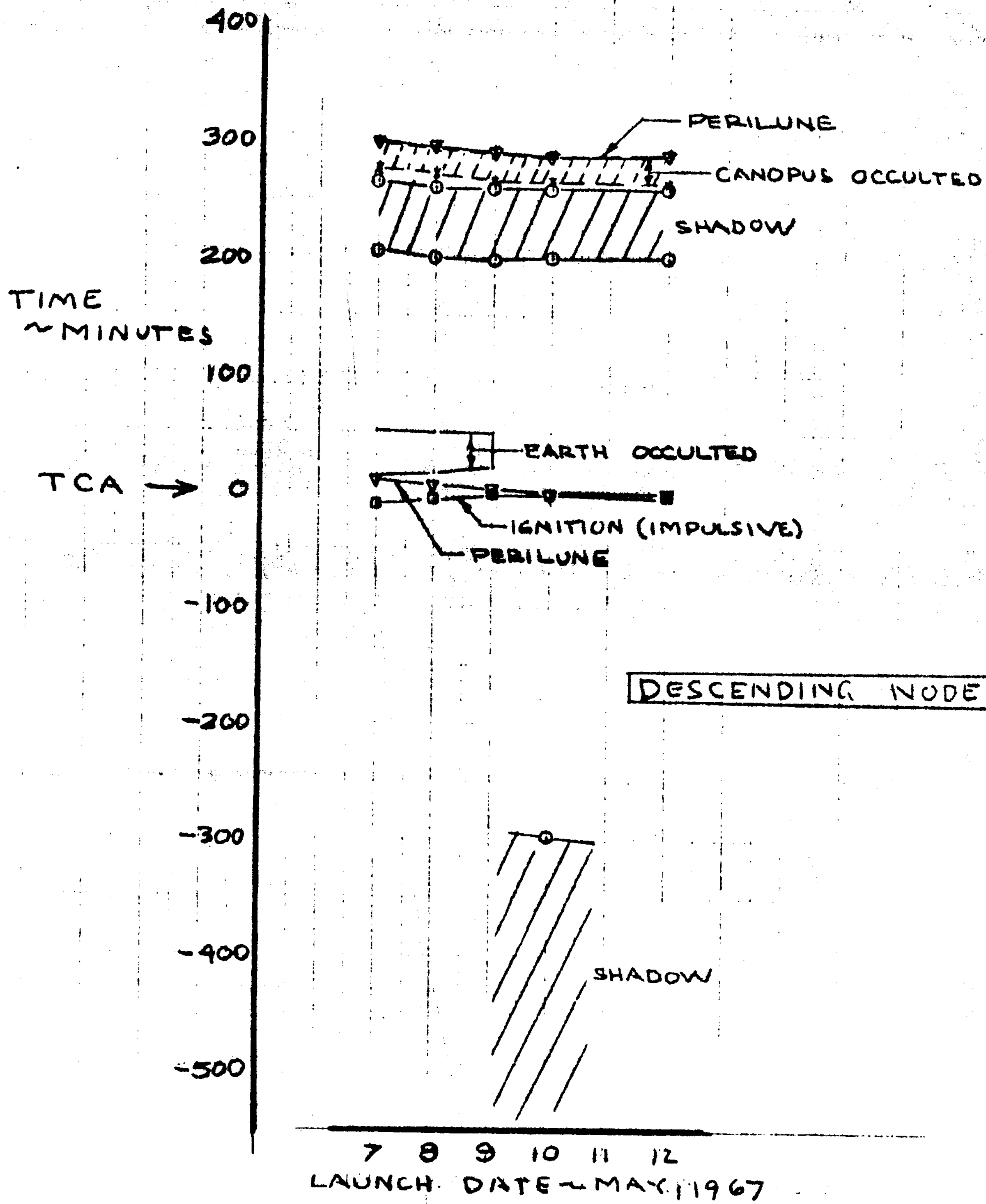
Fig. 4.4.0.2

TABLE I

MISSION DEFINITION

MISSION	CASE IB	CASE II C
Launch		
Date	May 9, 1967	July 3, 1967
Time (hr, min, sec)	21 57 29.2	18 56 25.3
Azimuth (deg)	90	90
P.O. Coast (sec)	2210.4	2154.3
Translunar Injection		
Time (hr, min, sec)	22 44 31.6	19 42 31.6
Latitude (deg)	- 28.19	- 27.94
Longitude (deg)	81.85	77.77
Translunar		
Transit Time (hr)	90	90
$\overline{B}\cdot\overline{T}$ (km)	4407	3819
$\overline{B}\cdot\overline{R}$ (km)	- 3345	- 3979
Approach Perilune Alt. (km)	242	271
Lunar Injection		
Date	May 13, 1967	July 7, 1967
Time (hr, min, sec)	18 42 17.8	13 37 33.6
Latitude (deg)	28.11	40.05
Longitude (deg)	65.90	57.32
Altitude (km)	257	334
Plane Change (deg)	7.17	8.22
Δ V (meters/sec)	587.3	609.6
Initial Orbit		
Apolune Altitude (km)	3000	3000
Perilune Altitude (km)	250	250
Inclination (deg)	- 33 (descending)	- 43 (descending)
Perilune Latitude (deg)	25.7	26.00
Perilune Longitude at Arrival (deg)	75.15	90.11
Transfer to Final Orbit		
Date	May 20, 1967	July 9, 1967
Latitude (deg)	- 24	- 26
Longitude (deg)	170	243.76
Δ V (meters/sec)	22.8	29.7
Final Orbit		
Apolune Altitude (km)	3000	3000
Perilune Altitude (km)	92	46
Inclination (deg)	- 33 (descending)	- 43 (descending)
Perilune Latitude (deg)	24	26
Perilune Longitude at Transfer (deg)	- 10	63.76
Commence Experiment		
Date	May 22, 1967	July 10, 1967

Fig. 4.4.0.3



	INITIALS	DATE	REV BY INITIALS	DATE	TITLE	MODEL
CALC	T HANSEN	7/27/5	RER	7/30	SEQUENCE OF EVENTS AT ARRIVAL MISSION IB	
CHECK						
APPD						
APPD						

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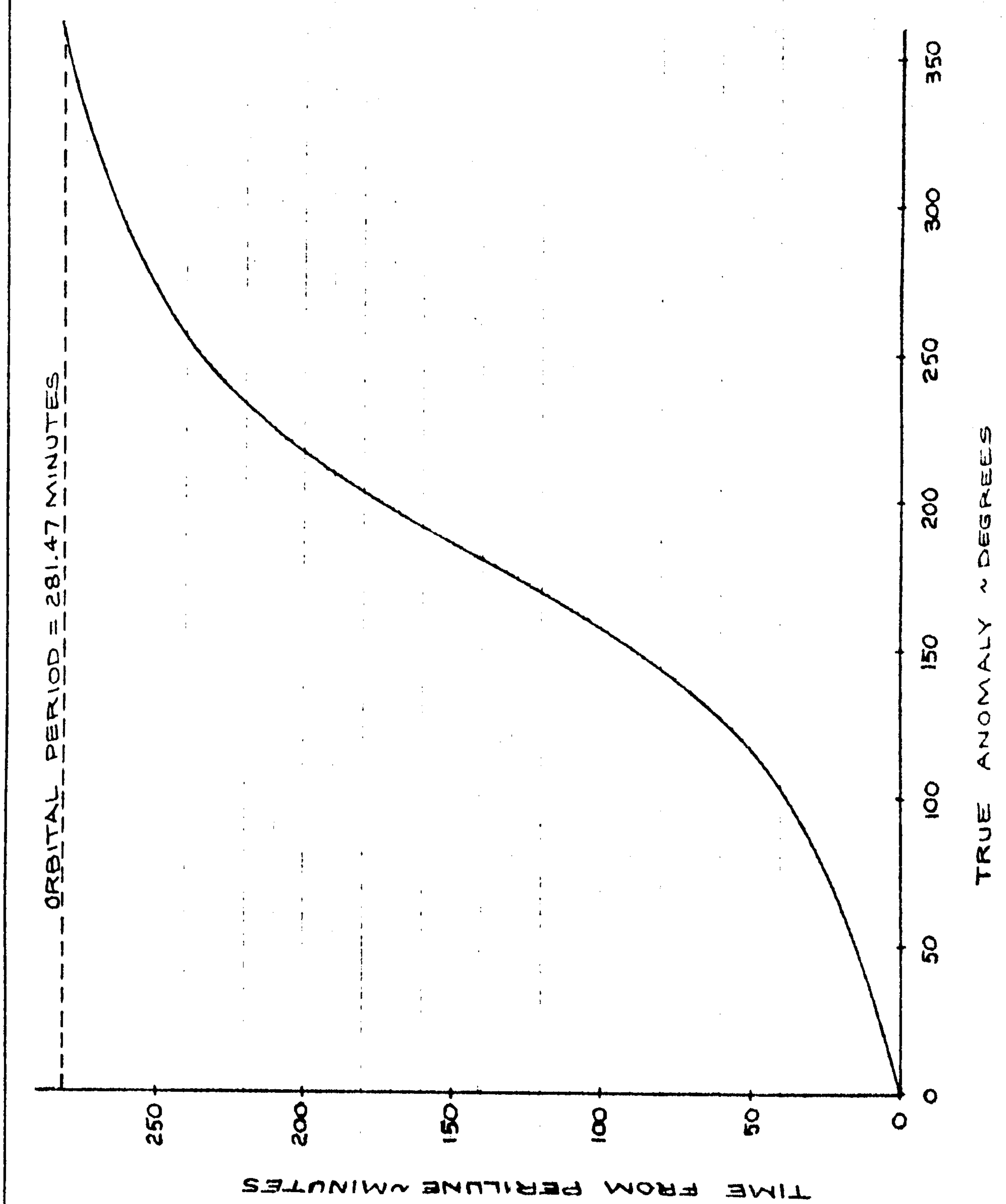
REV LTR

Fig. 4.4.0.4

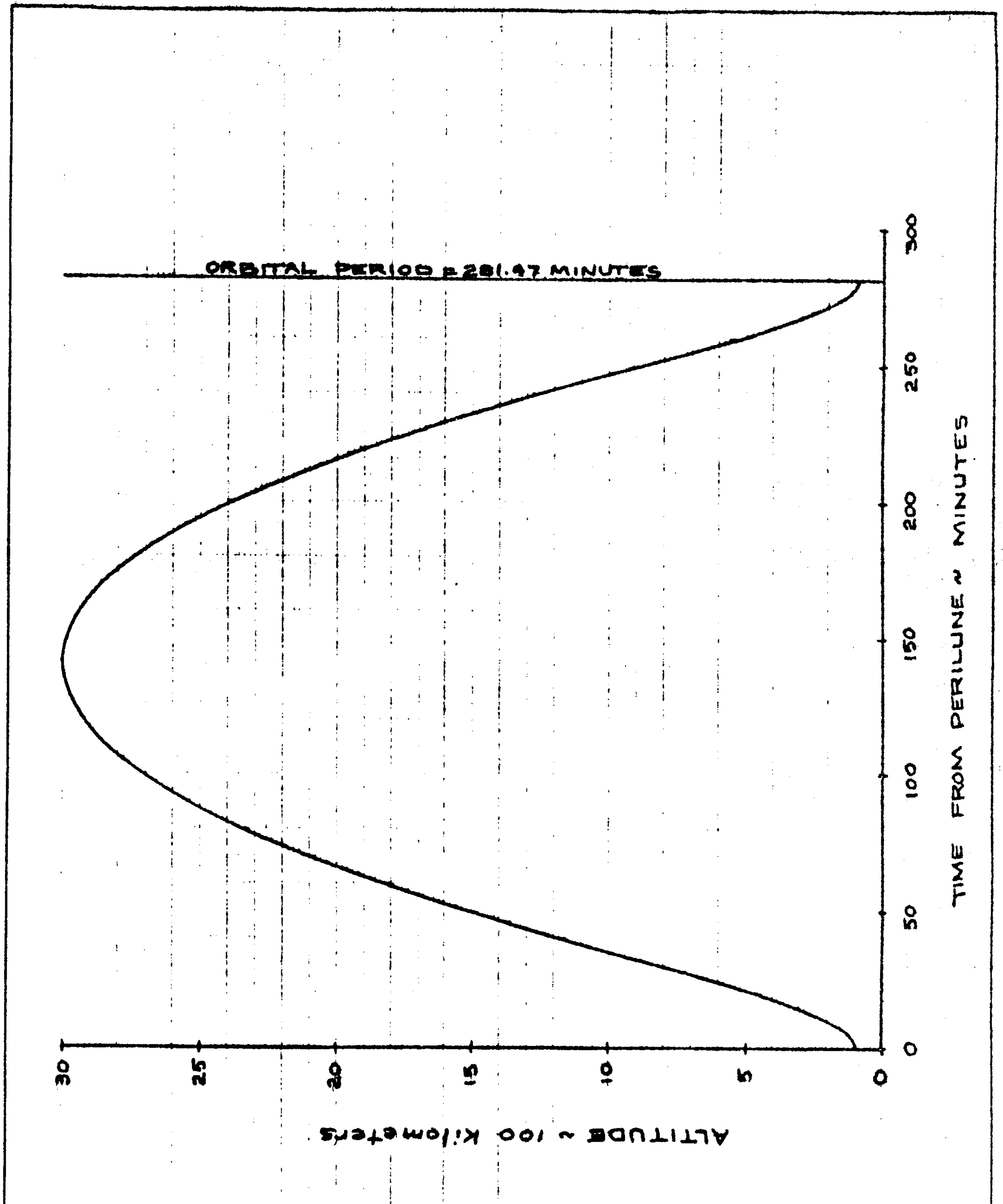


	INITIALS	DATE	REV BY INITIALS	DATE	TITLE	MODEL
CALC	J.F.B.	7/15/5			TIME-TRUE ANOMALY RELATIONSHIP (3000-92 Km. ORBIT) CASE I B	
CHECK						
APPD.						
APPD						

U3 4013 8000 REV. 12-64

Fig. 4.4.0.5

REV LTR________

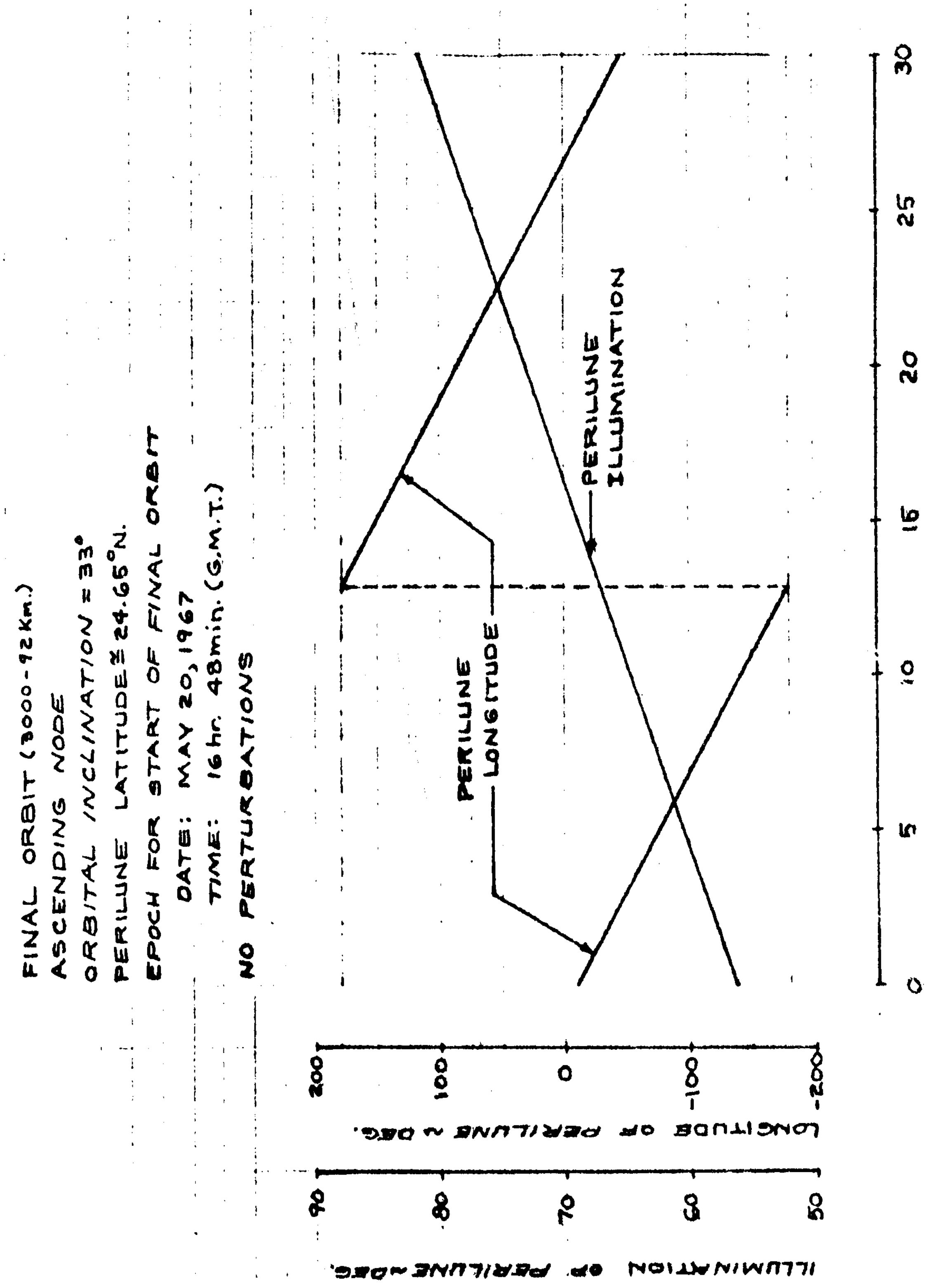


	INITIALS	DATE	REV BY INITIALS	DATE	TITLE	MODEL
CALC	J. F. B.	6/30/65			ALTITUDE AS A FUNCTION OF TIME FROM PERILUNE (3000 – 92 Km. ORBIT) CASE IB	
CHECK						
APPD.						
APPD.						

U3 4013 8000 REV. 12-64

Fig. 4.4.0.6

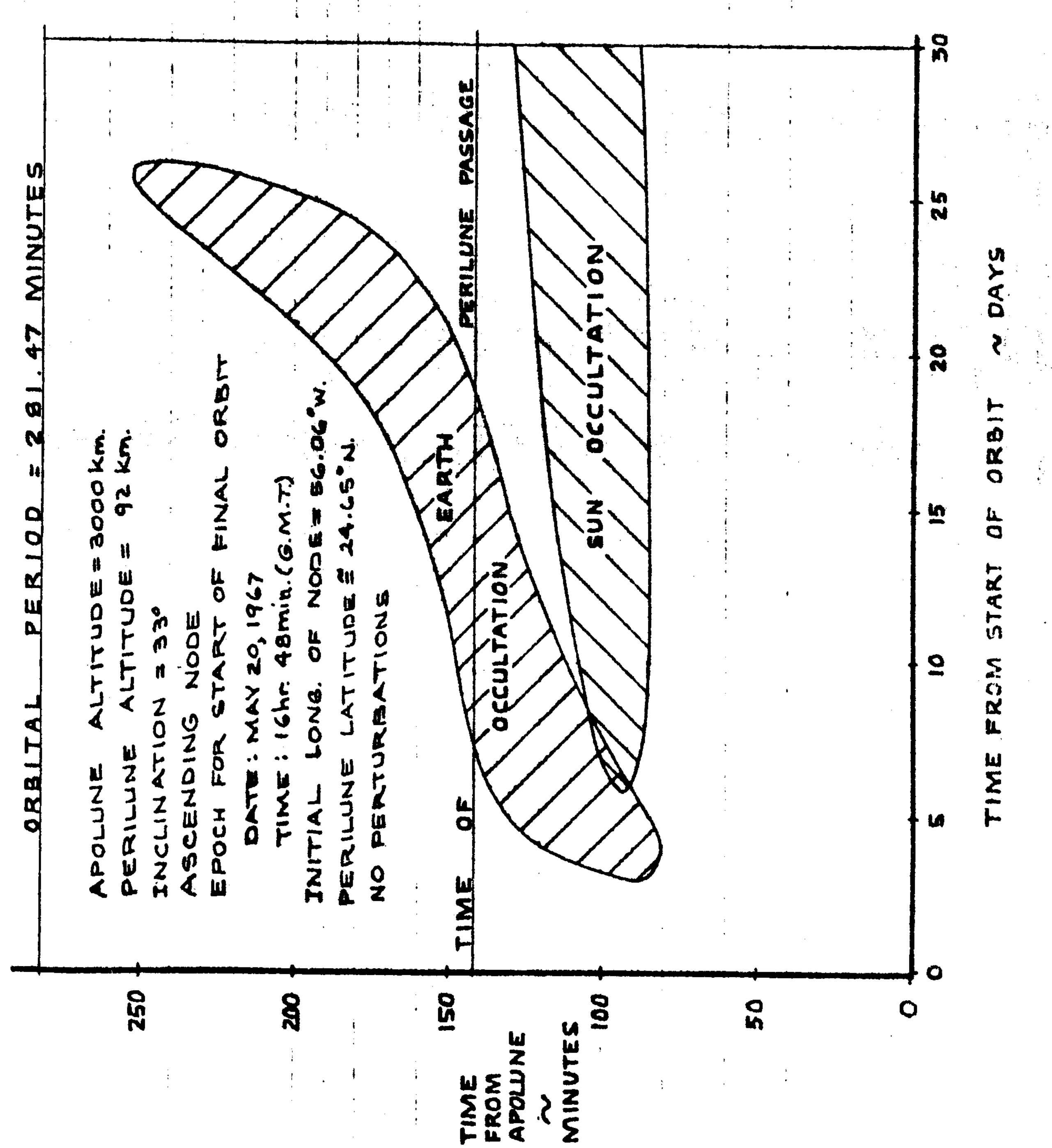
REV LTR__________



	INITIALS	DATE	REV BY INITIALS	DATE	TITLE	MODEL
CALC	J.F.B.	7/6/5			ILLUMINATION AND LONGITUDE OF PERILUNE IN FINAL ORBIT SCIENTIFIC EXP. CASE I B	
CHECK						
APPD.						
APPD.						

U3 4013 8000 REV. 12-64

Fig. 4.4.0.7

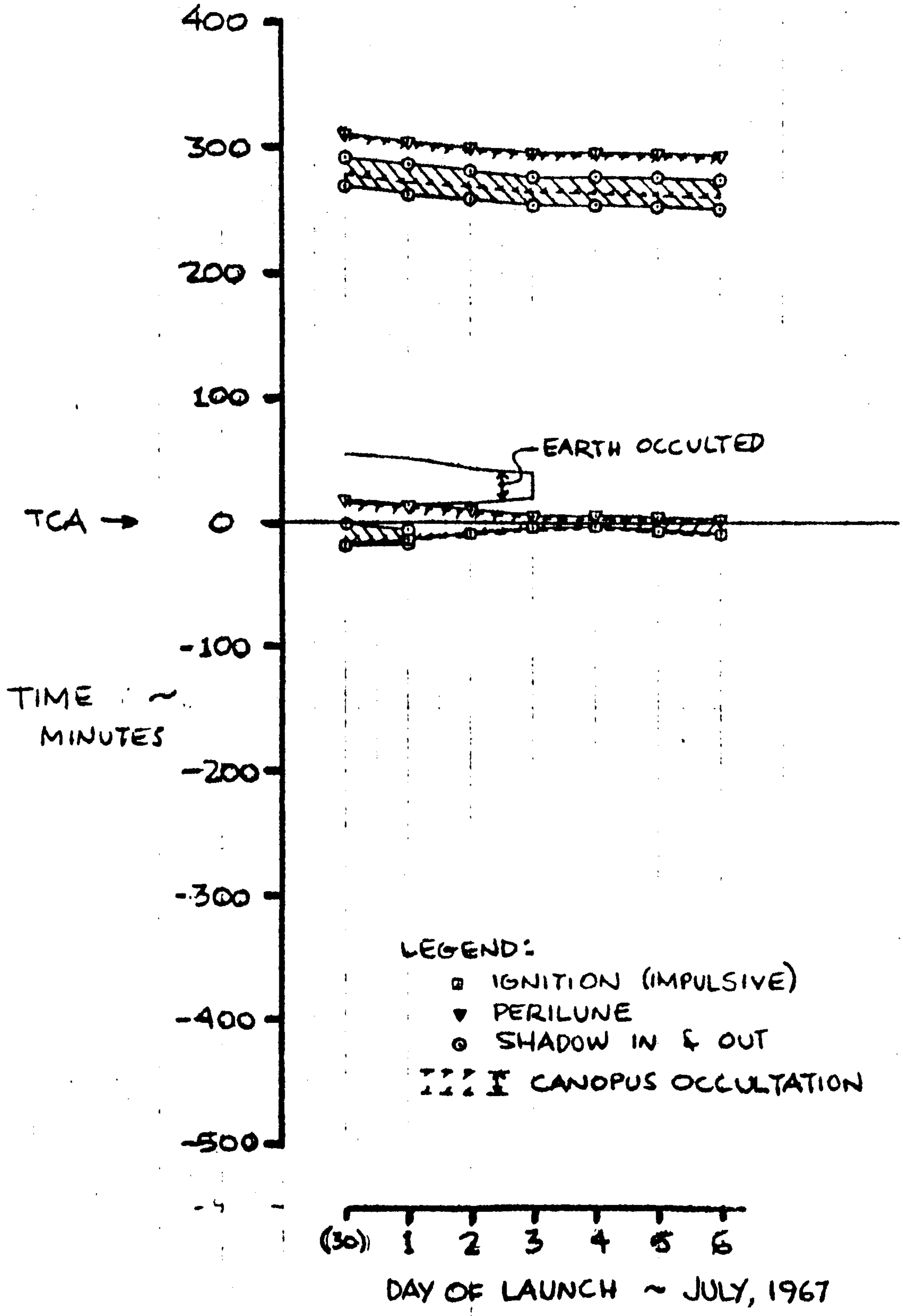


	INITIALS	DATE	REV BY INITIALS	DATE	TITLE	MODEL
CALC	J.F.B.	7/1/5			SUN AND EARTH OCCULTATION DURING FINAL LUNAR ORBIT SCIENTIFIC EXP. CASE I B	
CHECK						
APPD.						
APPD.						

U3 4013 8000 REV. 12-64

Fig. 4.4.0.8

REV LTR________

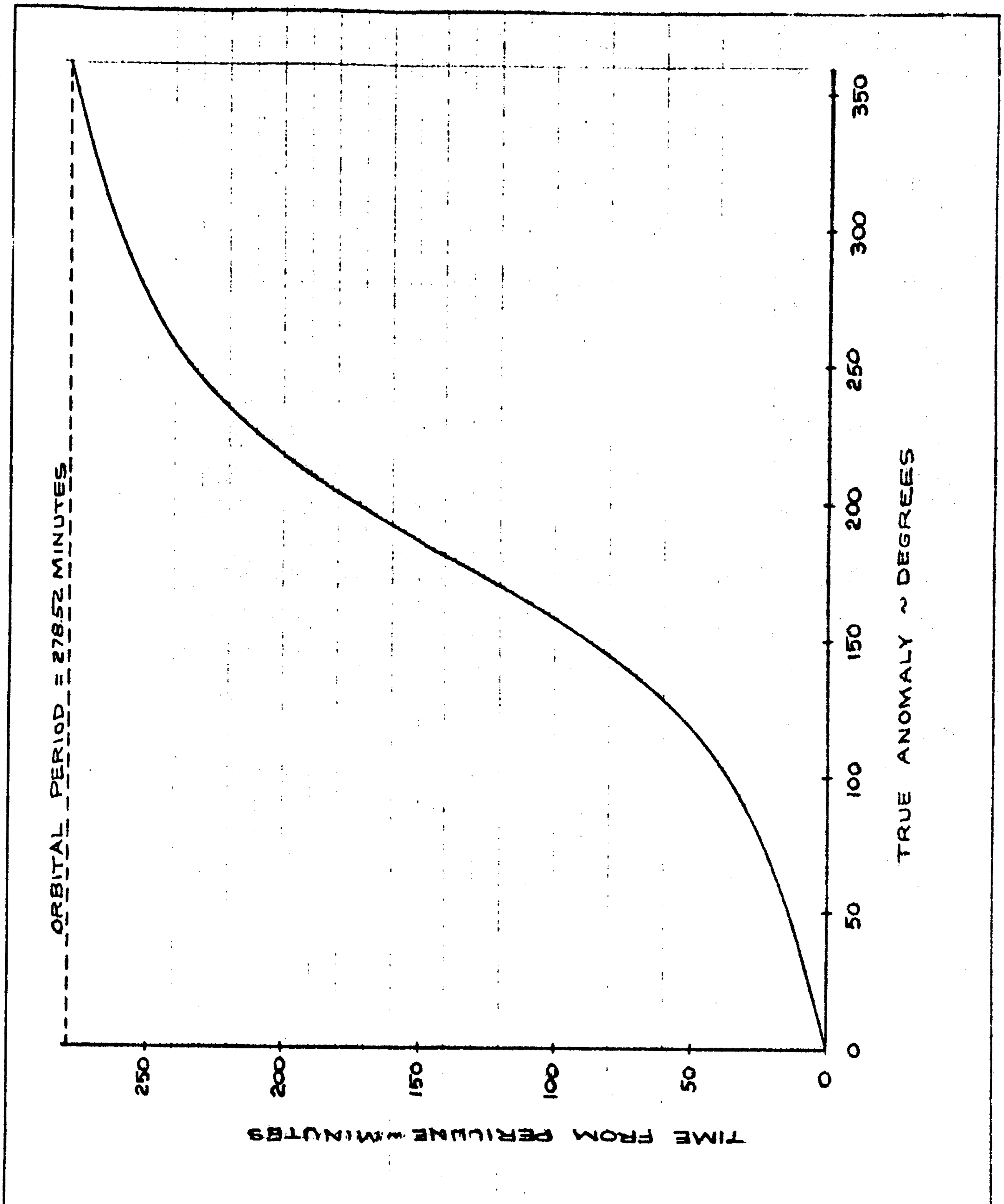


	INITIALS	DATE	REV BY INITIALS	DATE	TITLE	MODEL
CALC					SEQUENCE OF EVENTS AT ARRIVAL L/O SCIENTIFIC MISSION IIC	
CHECK						
APPD.						
APPD.						

U3 4013 8000 REV. 12-64

Fig. 4.4.0.9

REV LTR

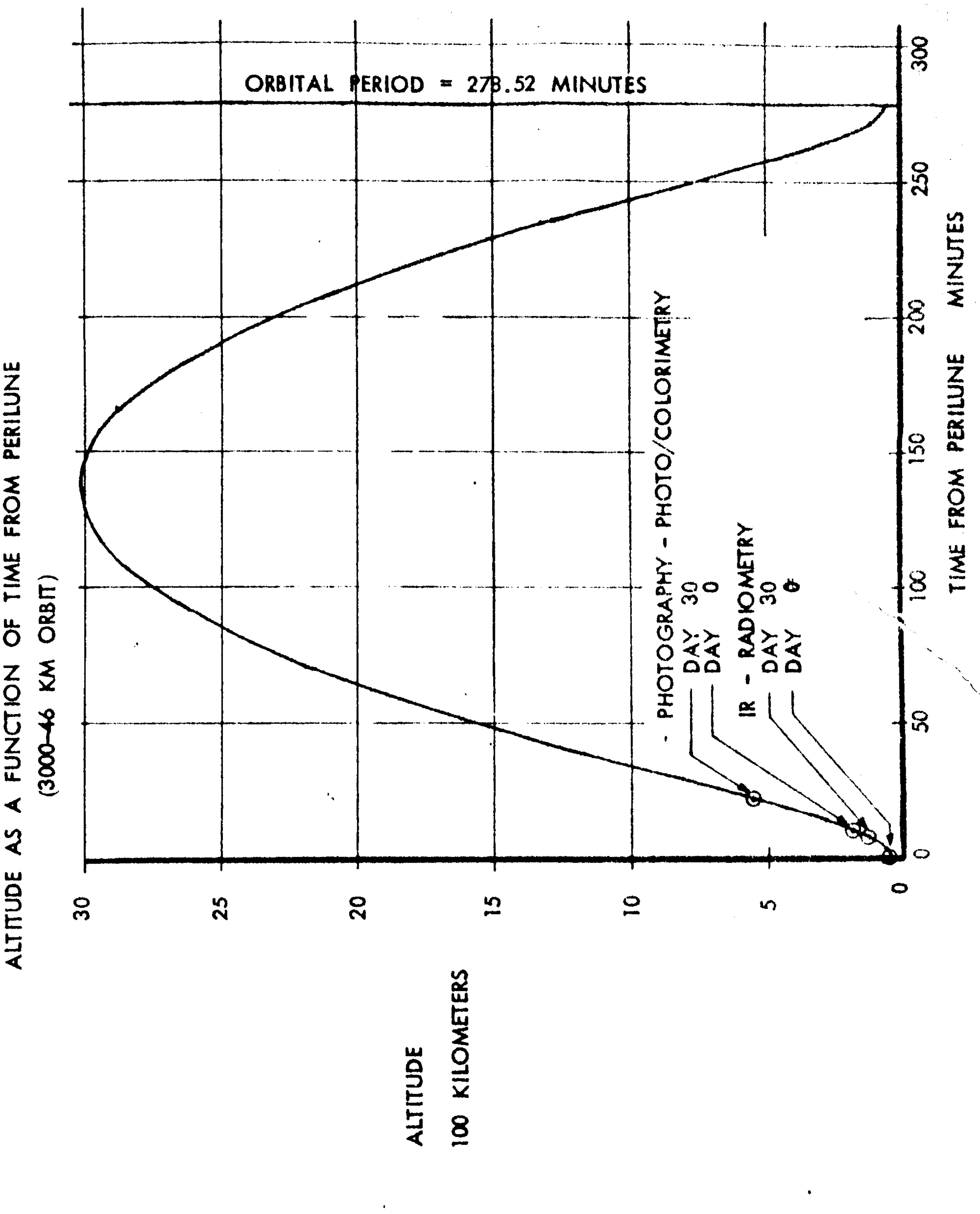


	INITIALS	DATE	REV BY INITIALS	DATE	TITLE	MODEL
CALC	J. F. B.	7/14/5			TIME – TRUE ANOMALY RELATIONSHIP (3000 – 46 Km. ORBIT) CASE II C	
CHECK						
APPD.						
APPD.						

U3 4013 8000 REV. 12-64

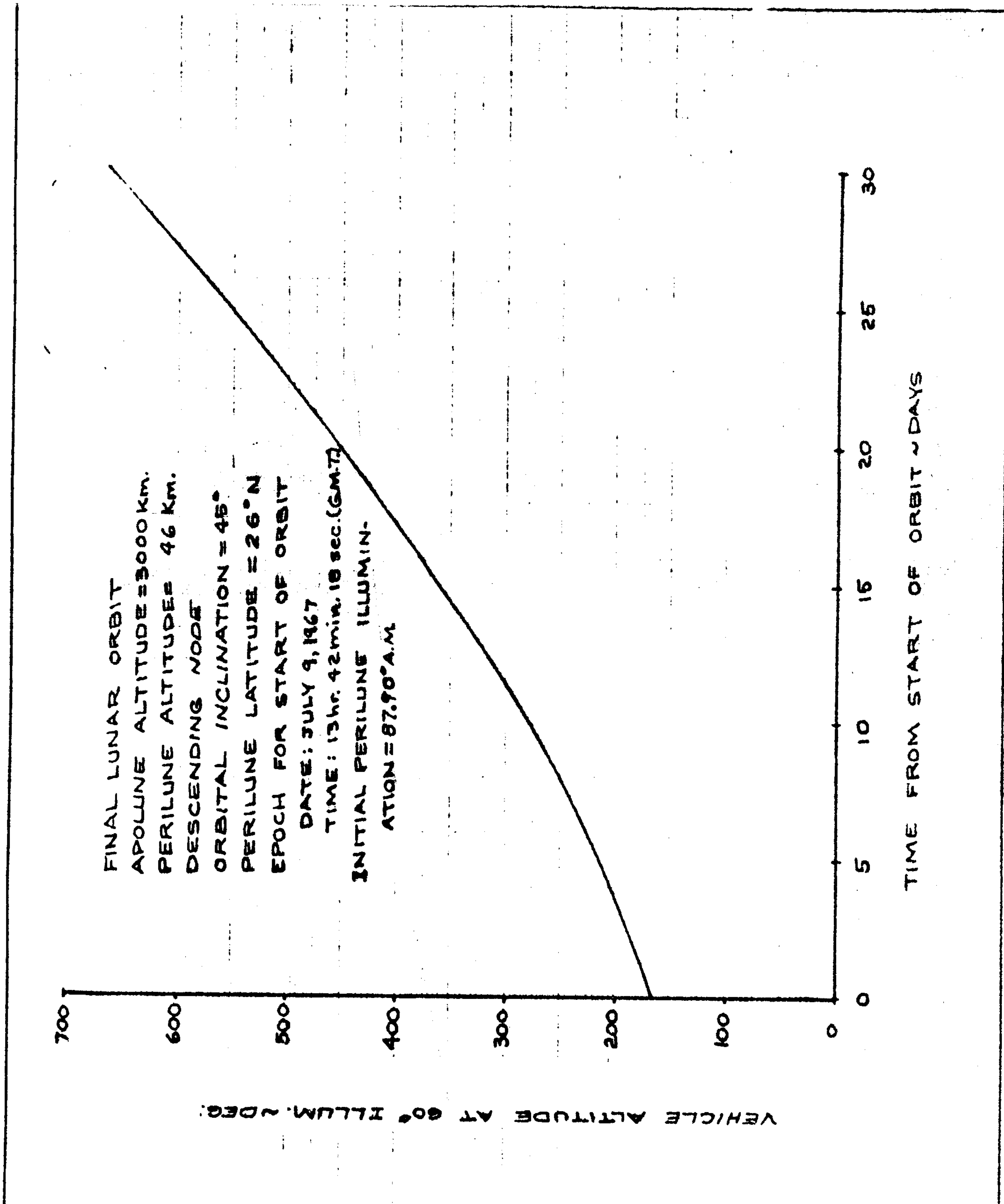
Fig. 4.4.0.10

REV LTR



ALTITUDE AS A FUNCTION OF TIME FROM PERILUNE
(3000-46 KM ORBIT)
ORBITAL PERIOD = 278.52 MINUTES
ALTITUDE
100 KILOMETERS
30
25
20
15
10
5
0
TIME FROM PERILUNE MINUTES
0
50
100
150
200
250
300
- PHOTOGRAPHY - PHOTO/COLORIMETRY
DAY 30
DAY 0
IR - RADIOMETRY
DAY 30
DAY 0

Fig. 4.4.0.11

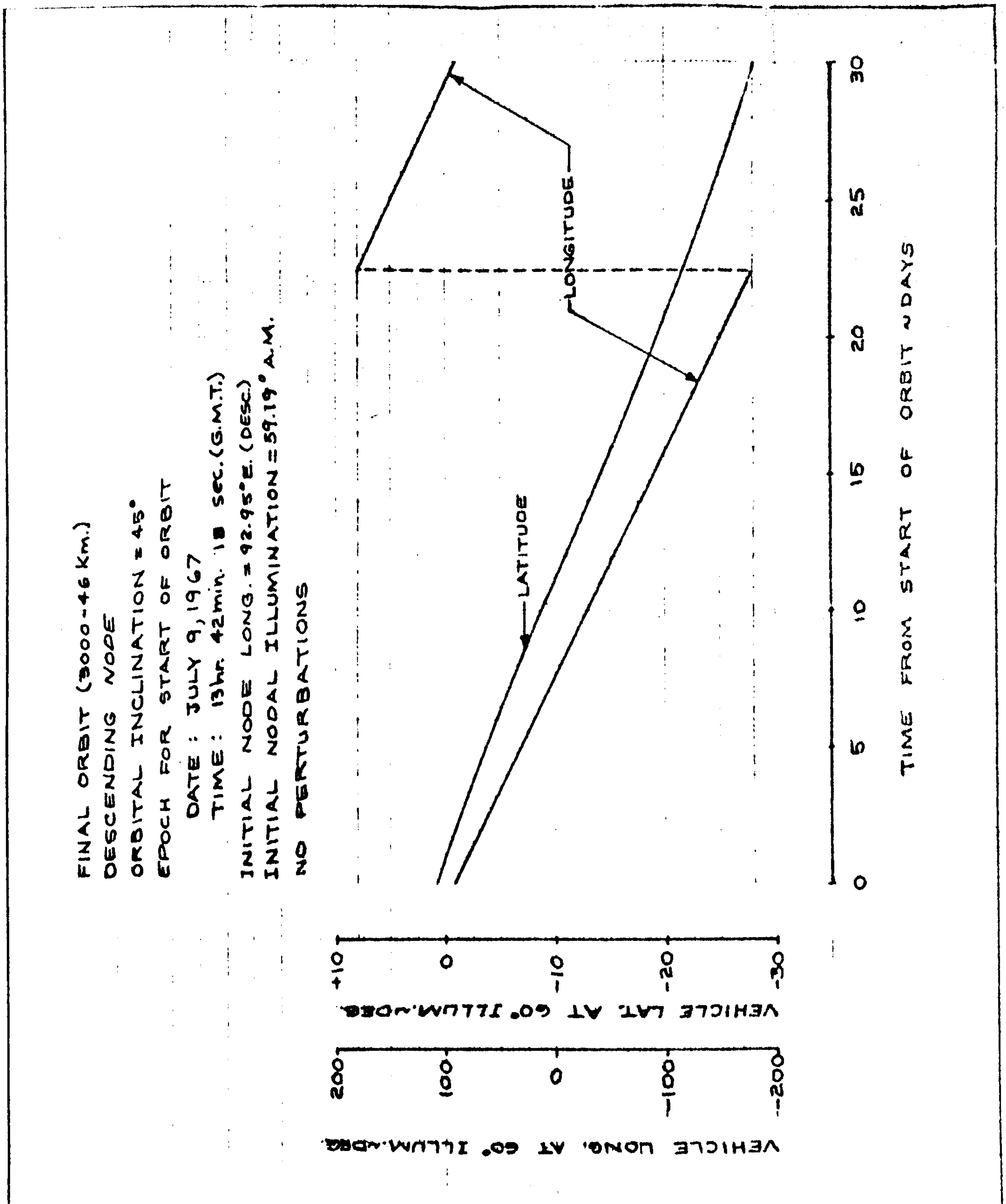


	INITIALS	DATE	REV BY INITIALS	DATE	TITLE	MODEL
CALC	J.F.B.	7/7/5			ALTITUDE OF VEHICLE AT 60° ILLUMINATION SCIENTIFIC EXP. CASE II C	
CHECK						
APPD.						
APPD.						

U3 4013 8000 REV. 12-64

Fig. 4.4.0.12

REV LTR________

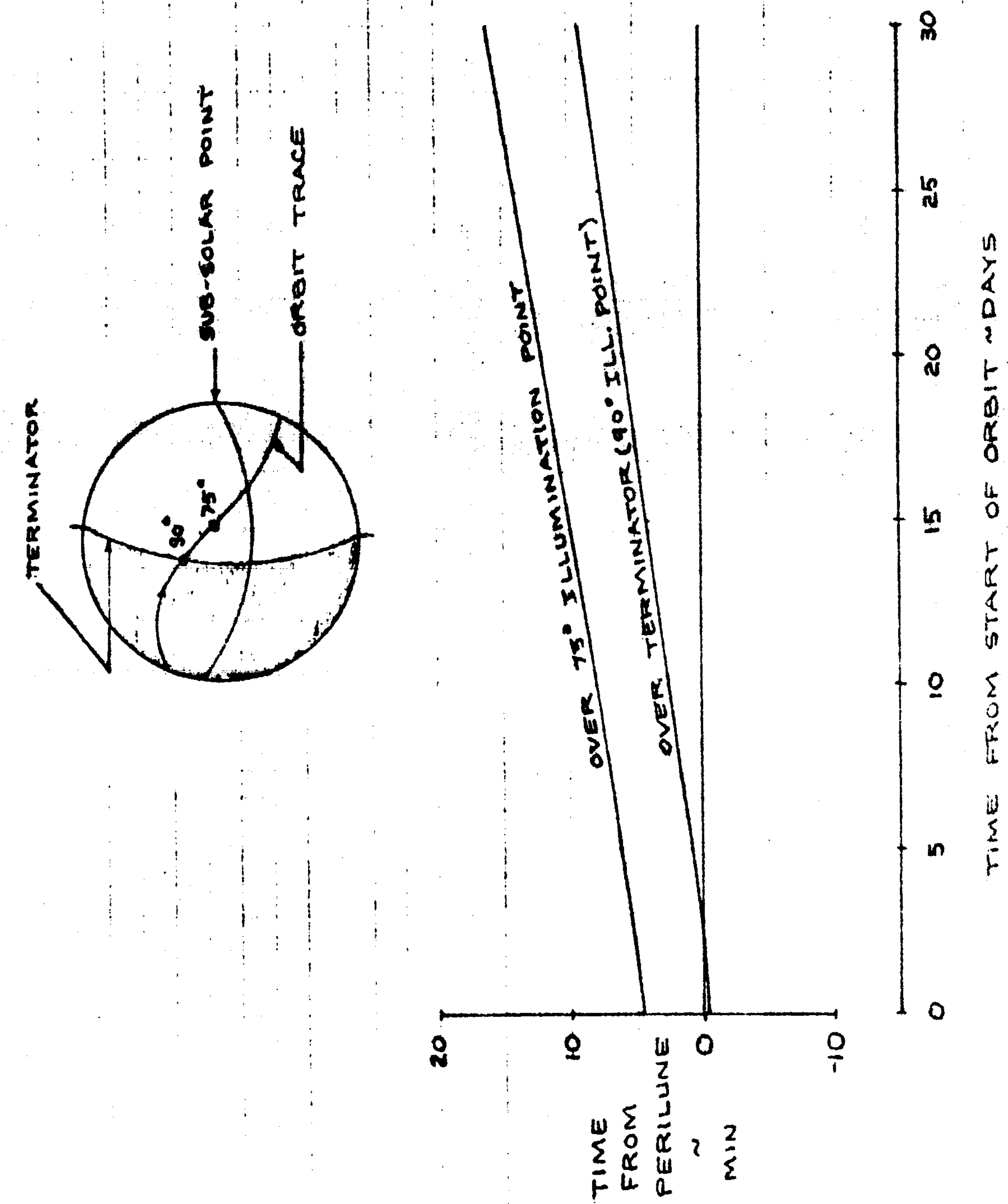


	INITIALS	DATE	REV BY INITIALS	DATE	TITLE	MODEL
CALC	J.F.B.	7/7/65			LATITUDE AND LONGITUDE OF VEHICLE AT 60° ILLUMINATION SCIENTIFIC EXP. CASE II C	
CHECK						
APPD.						
APPD.						

U3 4013 8000 REV. 12-64

Fig. 4.4.0.13

REV LTR______

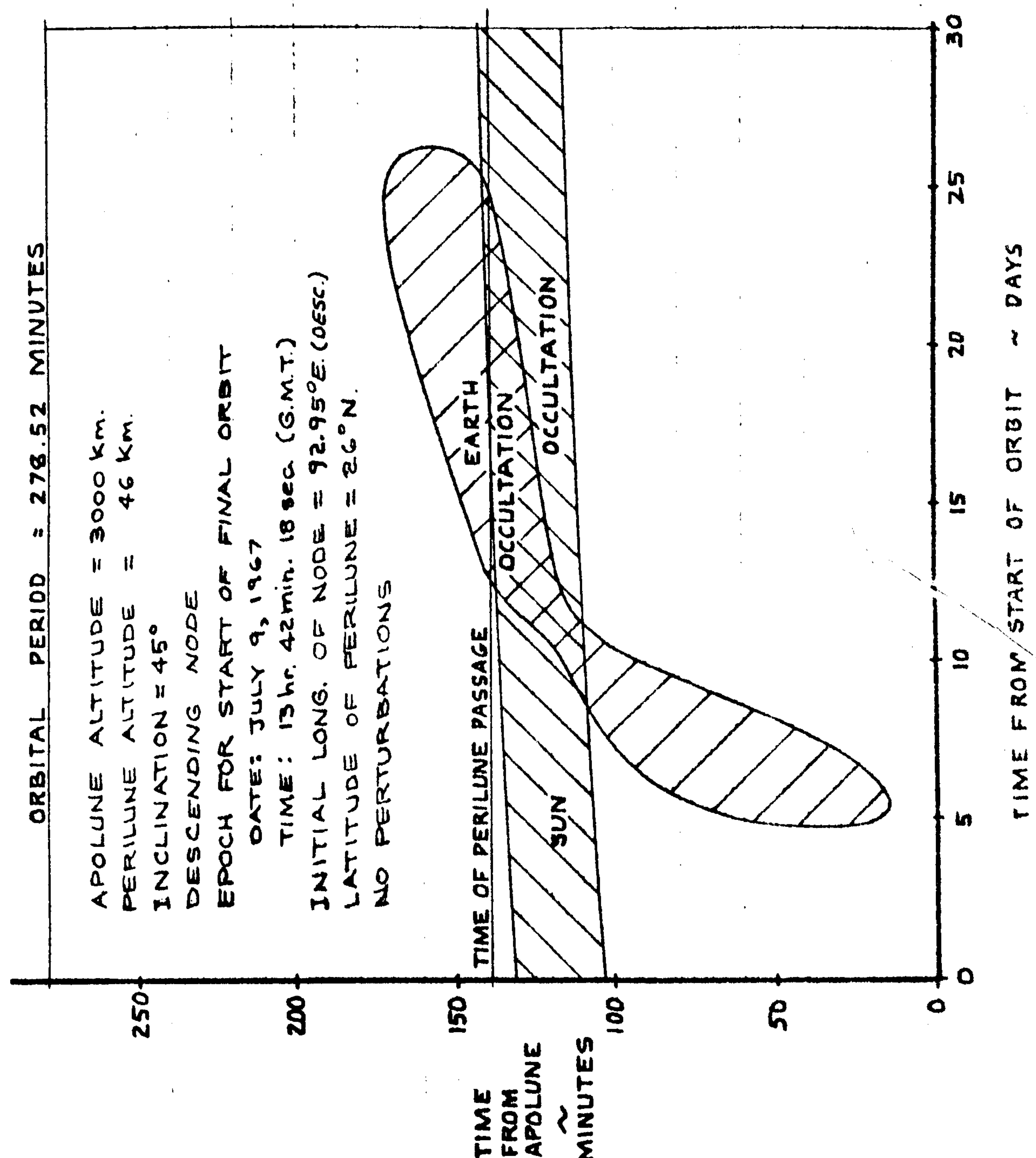


	INITIALS	DATE	REV BY INITIALS	DATE	TITLE	MODEL
CALC	J. F. B.	7/15/5			TIMES OF VEHICLE PASSAGE OVER 75° AND 90° ILLUMINATION POINTS SCIENTIFIC EXP. CASE II C	
CHECK						
APPD.						
APPD.						

U3 4013 8000 REV. 12-64

Fig. 4.4.0.14

REV LTR________



	INITIALS	DATE	REV BY INITIALS	DATE	TITLE	MODEL
CALC	RAU	7/2/65			SUN AND EARTH OCCULTATION DURING FINAL LUNAR ORBIT SCIENTIFIC EXP. CASE II C	
CHECK	J.F.B.	7/7/6				
APPD.						
APPD.						

U3 4013 8000 REV. 12-64

Fig. 4.4.0.15

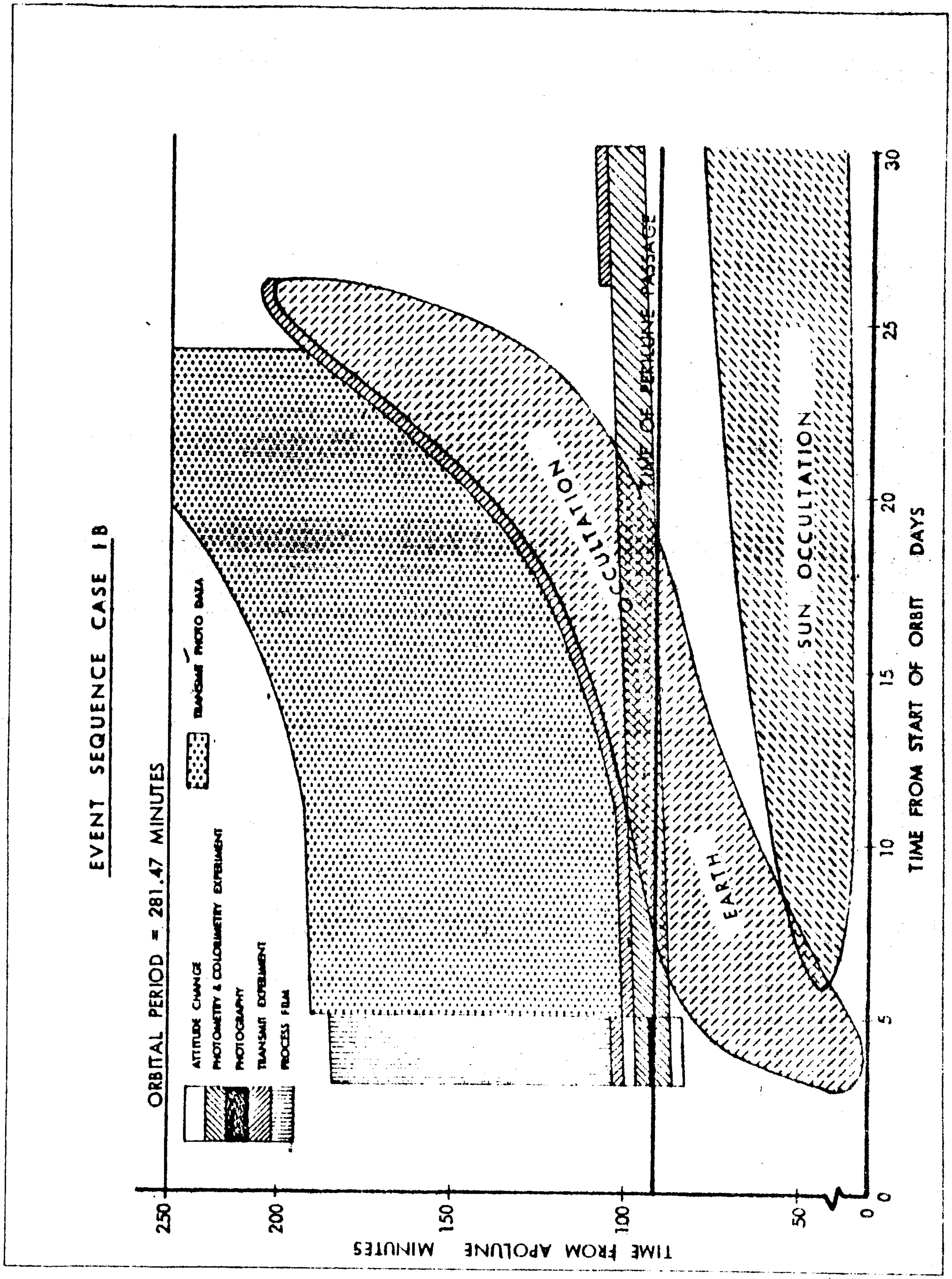
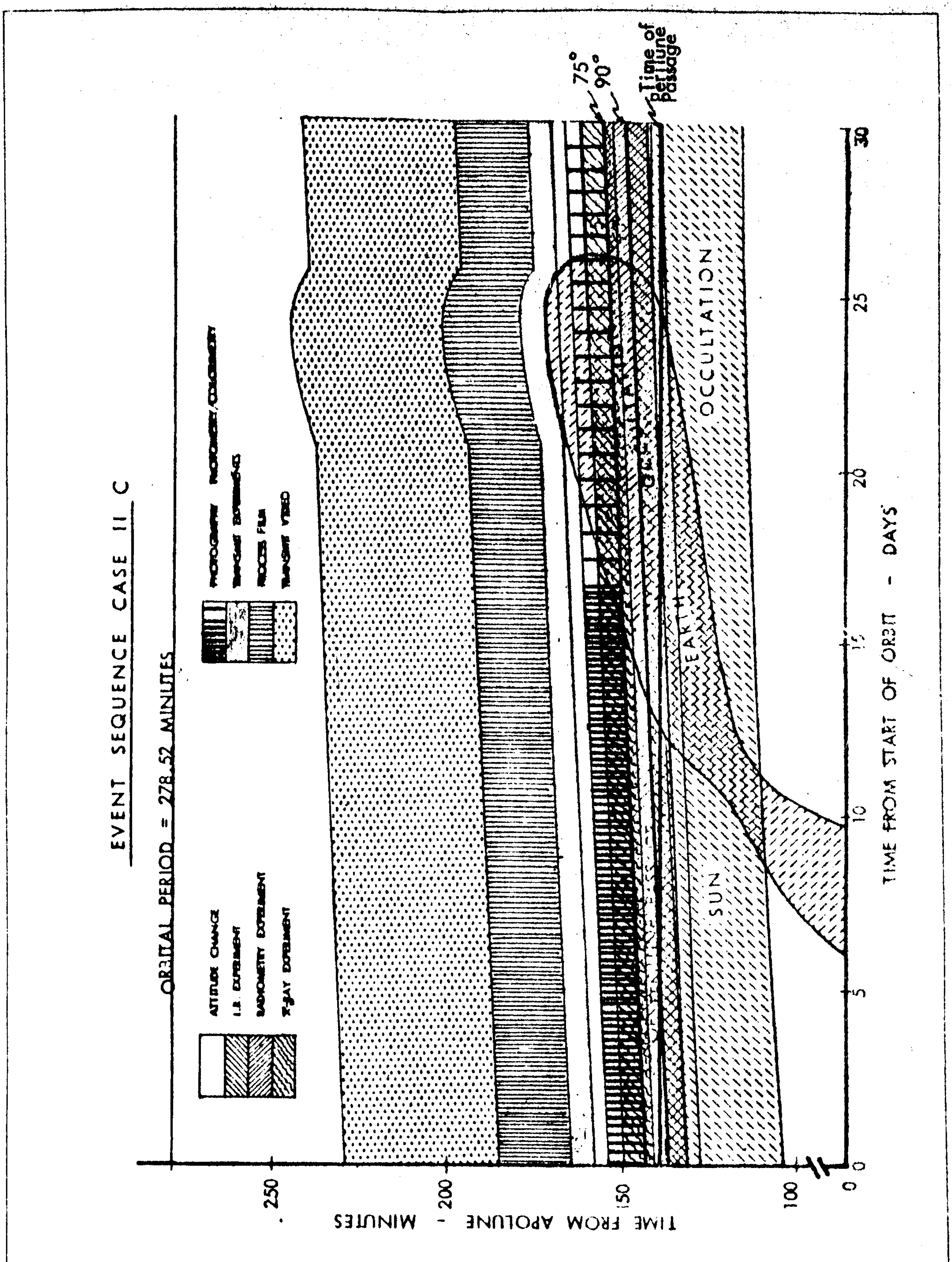

EVENT SEQUENCE CASE IB
ORBITAL PERIOD = 281.47 MINUTES
ATTITUDE CHANGE
PHOTOMETRY & COLORIMETRY EXPERIMENT
PHOTOGRAPHY
TRANSMIT EXPERIMENT
PROCESS FILM
TRANSMIT PHOTO DATA
EARTH OCCULTATION
SUN OCCULTATION
TIME OF PERILUNE PASSAGE
TIME FROM APOLUNE MINUTES
250
200
150
100
50
0
TIME FROM START OF ORBIT DAYS
0
5
10
15
20
25
30

Fig. 4.4.0.16



EVENT SEQUENCE CASE II C
ORBITAL PERIOD = 278.52 MINUTES
ATTITUDE CHANGE
I.R. EXPERIMENT
RADIOMETRY EXPERIMENT
X-RAY EXPERIMENT
PHOTOGRAPHY PHOTOMETRY/COLORIMETRY
TRANSIENT EXPERIMENTS
PROCESS FILM
TRANSMIT VIDEO
75°
90°
Time of Perilune Passage
EARTH OCCULTATION
SUN
0
5
10
15
20
25
30
TIME FROM START OF ORBIT - DAYS
0
100
150
200
250
TIME FROM APOLUNE - MINUTES

Fig. 4.4.0.17

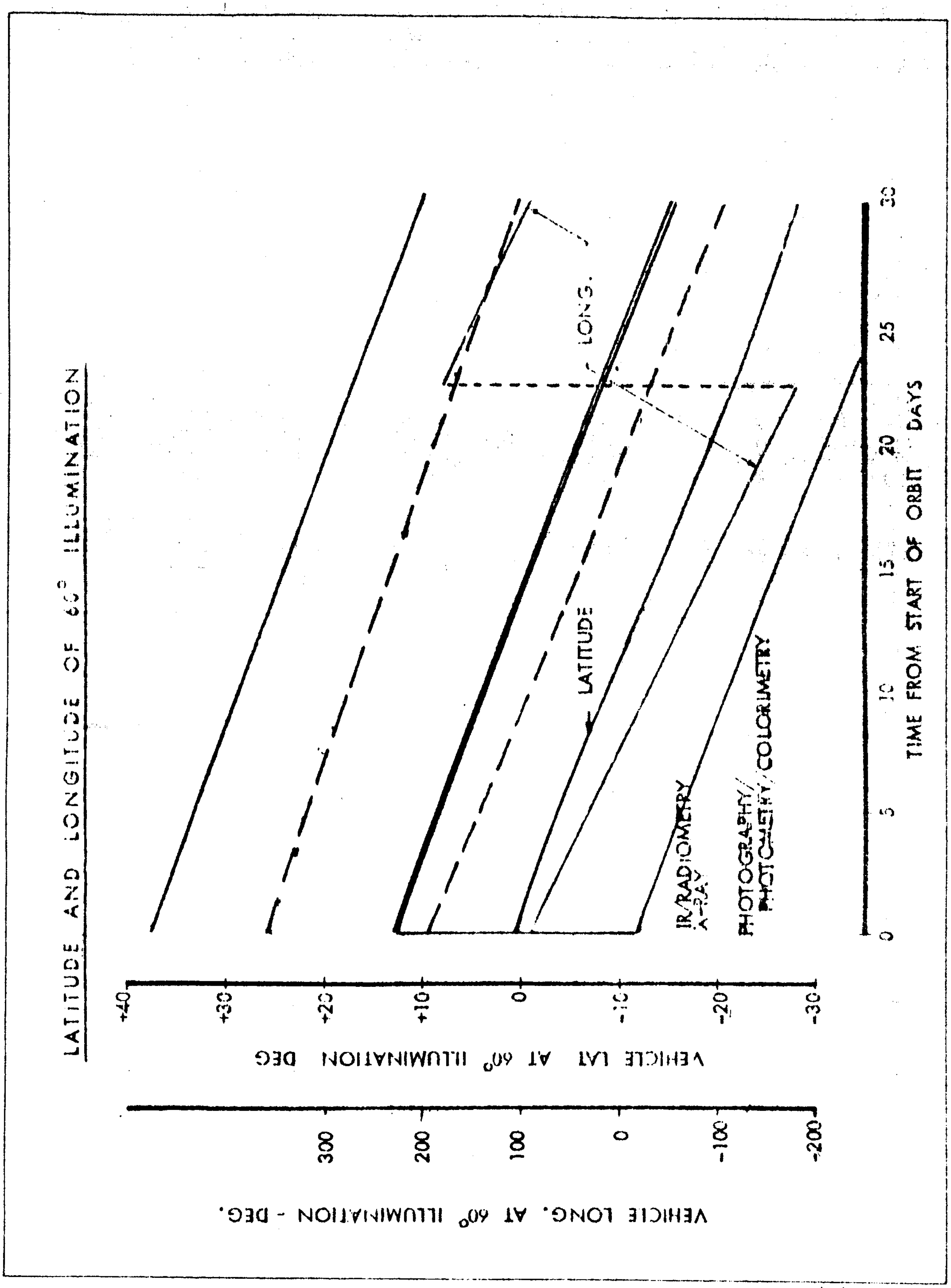

LATITUDE AND LONGITUDE OF 60° ILLUMINATION
VEHICLE LAT AT 60° ILLUMINATION DEG
+40
+30
+20
+10
0
-10
-20
-30
VEHICLE LONG. AT 60° ILLUMINATION - DEG.
300
200
100
0
-100
-200
TIME FROM START OF ORBIT DAYS
0
5
10
15
20
25
30
LONG.
LATITUDE
IR/RADIOMETRY
X-RAY
PHOTOGRAPHY/
PHOTOMETRY/COLORIMETRY

Fig. 4.4.0.18

4.4.1 (Continued)

assured by experiment scan system design, such that a 43 km crossrange strip is covered from an altitude of 100 km, in this case.

It is to be noted, by reference to Figures 4.4.0.5 and 4.4.0.6, that under the above definition of the photometry/colorimetry experiment the altitude of the experiment will change continuously over the duration of the mission. This is due to the precession of the earth system around the sun while the lunar orbit remains approximately fixed in inertial space. As a result the illumination band passes relative to the perilune of the orbit at a rate of 360/365 degrees per day. The photometry/colorimetry experiment, assumed to be performed in a fixed illumination band, is therefore executed at a progressively increasing true anomally and a correspondingly increasing altitude. This is an arbitrary definition, for illustration purposes only, since the experiment could be performed equally well from a nearly constant altitude at progressively changing illumination angles (1°/day), which may be preferable from the scientific viewpoint, up to the limit of the perilune approaching the terminator.

It should also be noted, with respect to Figure 4.4.0.16 that the event sequence layout was performed on a fixed experiment duration basis without an attempt at optimization of coverage, or detailed coverage definition. This was done in the interests of simplicity of presentation since the only purpose of the event sequence lyaout in the preliminary investigation phase is to provide data for an initial subsystem analysis. Detailed time analyses will have to be performed at

4.4.1 (Continued)

the experiment requirement specifications. A compressed time scale of functions per orbital pass appears to be feasible, although not necessary, by elimination of dual time slot allocation for mutually exclusive functions.

4.4.2 CASE IIC

The mission profile for Case IIC (Medium Resolution Photo) prior to injection into the final lunar orbit will be analogous to the standard Lunar Orbiter mission. In the final orbit phase the surface oriented experiments will be performed at the time(s) when the spacecraft passes over the illumination band appropriate to a particular experiment or experiment set. It should be noted that the mission profile in this case is illustrative only. For example, the IR and Radiometry experiments are initiated prior to the passage over the terminator into the illuminated region of the surface. The IR readings are therefore taken over a region which had been in solar shadow for 14 days and represents the coldest condition. The equivalent case for the entry of the spacecraft into the shadowed region, corresponding to the sunset condition with respect to a surface point, can be constructed to be a reversal of the illustrated sequence of events (i.e. placement of photography/photometry/colorimetry as the first experiment in the sequence) under the assumption that the orbit perilune occurs at the sunset terminator.

The areas of latitude coverage for the experiments of Case IIC resulting from the event sequencing specified by Figure 4.4.0.17 over a period of 30 days (360° longitude coverage) are shown in Figure 4.4.0.18. The precession in latitude of the coverage levels is a result of the relative

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4.4.2 (Continued)

motion of the solar illumination bands, specified for the individual experiments, with respect to the inertially fixed orbit ignoring perturbations due to gravitational anomalies and earth effect. It should be noted that an overlap band between the various experiments exists. This can be utilized as an aid in definition of the surface area of observation in addition to the obvious means of extrapolating from photographic location points using orbital data over the relatively short arc length involved.

The experiment altitude will change as the mission progresses for reasons identical to these causing latitude precession. The variation of experiment altitude, due to precession of a given solar illumination band with respect to orbit perilune, over the 30 day mission is illustrated in Figure 4.4.0.11. It should be noted that, within limits, the experiment altitude can be held constant if the experiment solar illumination at the time of experiment operation is allowed to change. The latter is, similarly, true in the case of latitude precession of the coverage band. Mapping of a constant latitude band is possible under the assumption that changing solar illumination conditions are acceptable. With respect to Figure 4.4.0.17, showing the event sequence for Case IIC, it should be noted that sufficient time is available in a single orbital pass to perform the required functions under a non-optimum allocation of a constant time slot for mutually exclusive functions. Compression of functions into a narrower time band is feasible, if required, and should be done when a detailed functional definition of experiments becomes available.

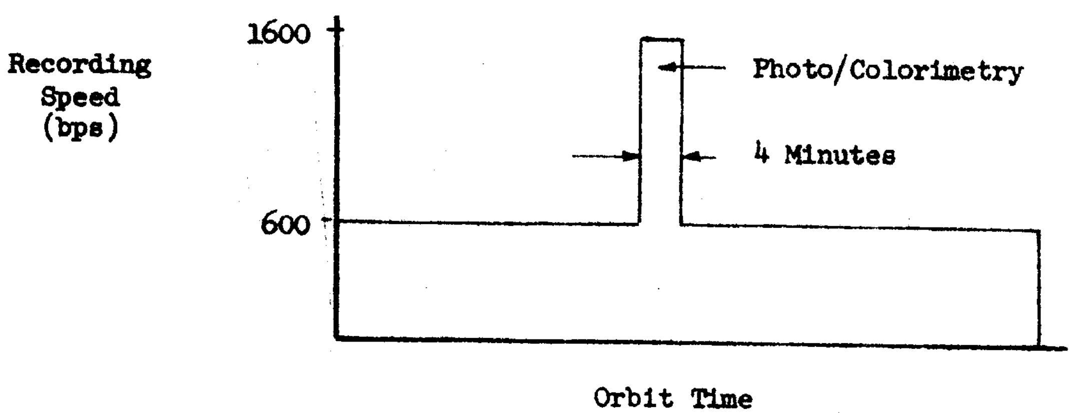
4.5.0 Subsystem Analyses

Subsystem analyses were carried out under the definitions of mission profiles and event sequences discussed in the preceding subsections. The results of these analyses in terms of performance identification of subsystem modification requirements, if any, and modification trade factors are discussed in the following paragraphs.

4.5.1 Communications Subsystem

For the purpose of determining the communication system configuration it has been assumed that the individual experiment sensors perform the necessary signal conditioning to supply to the communication system binary signals at the requisite data rates listed.

The Case IB Configuration, with the exception of the orbits during which the high resolution photographic data is being transmitted to earth, requires a data rate as shown in the figure below.



Data Rate as a Function of Orbit

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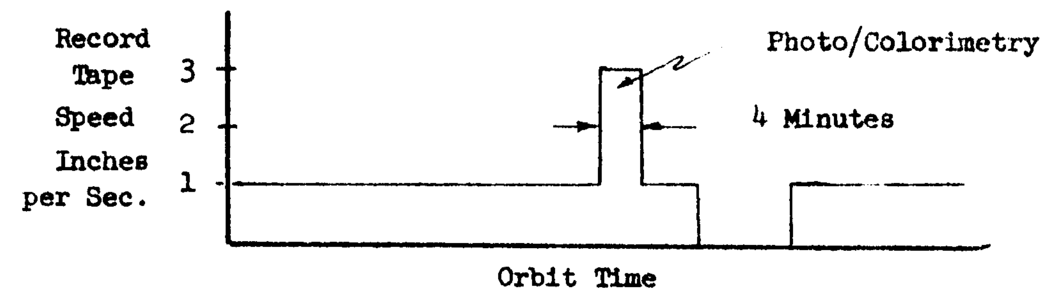
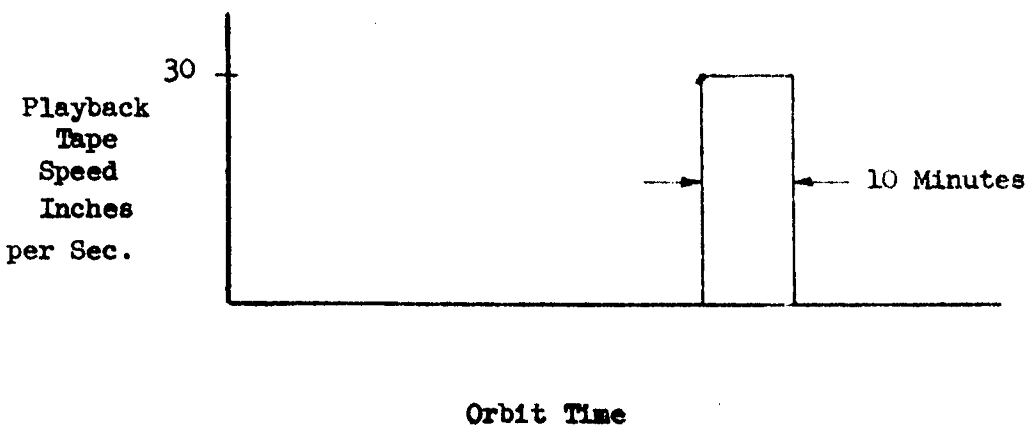
4.5.1 **Communications Subsystem (Cont.)**

The present Lunar Orbiter communication system is not capable of continuously transmitting at the above data rate, except in the high power mode, without a modification. It will be necessary, as discussed in section 3.2.5 to store the data on magnetic tape for a complete orbit and upon completion to transmit this data to earth at an increased rate. As this transmission would use the present video (Mode 2) system it requires only minor changes to the present Lunar Orbiter communication system.

Based on an investigation of capabilities of tape recorders designed for Deep Space operation, the recording and playback requirements as shown in the figures below can be met.

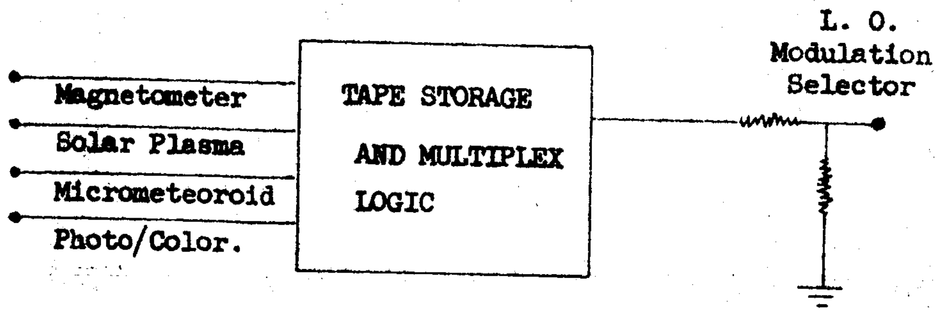




The configuration of the tape recorder relative to experiment inputs and the Lunar Orbiter modulation selector is shown in the figure below:

Communications Subsystem (Cont.)

Experiment Input



The high power transmission mode, on a time share basis with video data is assumed in the above configuration.

It should be noted that the above figure assumes a self contained multiplexing logic capability in the tape recorders. This capability is available within developmental models and can be adapted to particular requirements with minor logic modification. An example of such a recorder is the Leach MTR-2000.

The time multiplexer-encoder or commutator between the digital experiment outputs and the tape recorder and communication system input is required to handle, upon command, two data rate modes.

MODE I

Inputs:	Micrometeroid	-- 4 bps
	Solar Plasma	-- 500 bps
	Magnetometer	-- 100 bps
	Spacecraft Time	-- 50 bps

Output Data Rate = 650 bps

Operation: Parallel to Serial

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MODE II

Inputs:	Micrometeroid	--	4 bps
	Solar Plasma	--	500 bps
	Magnetometer	--	100 bps
	Spacecraft Time	--	50 bps
	Photometry/Colorimetry	--	1000 bps

Output Data Rate = 1700 bps

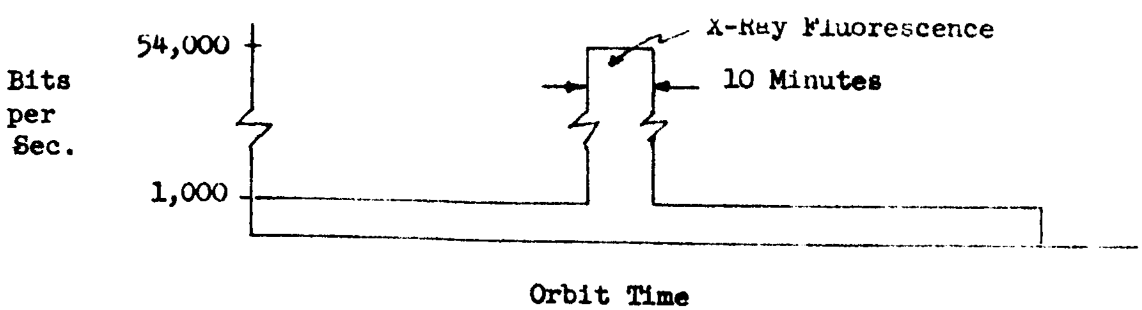
Operation: Parallel to Serial

Mode II differs from Mode I in that it has the additional experiment, photometry/colorimetry (1000 bps), to multiplex into the single output data rate. This change in data rate will have to be achieved by preprogrammed command.

The above requirements, in conjunction with the commutation of data over the entire orbital period, result in the following tape recorder specification:

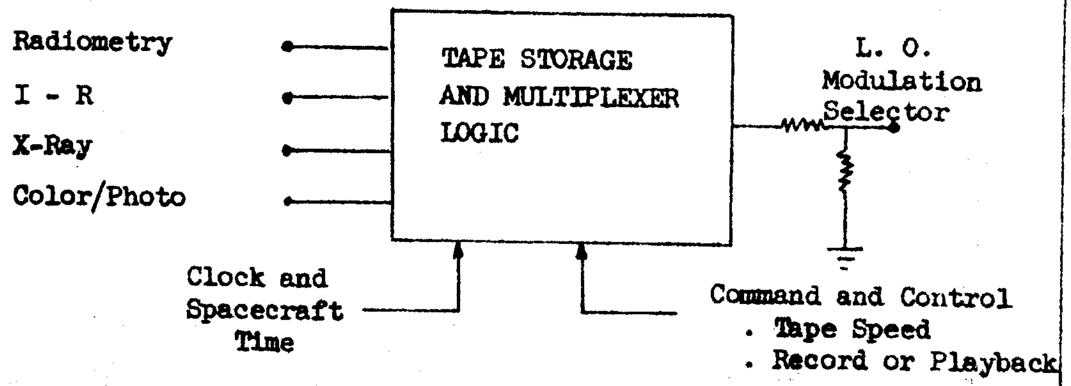
Recording speed	-- 1 and 3/inches/second
Playback speed	-- 30/inches/second
Tape length	2000 feet
Tape density	700 bits/second

The data rate profile for Case IIC is shown in the figure below:

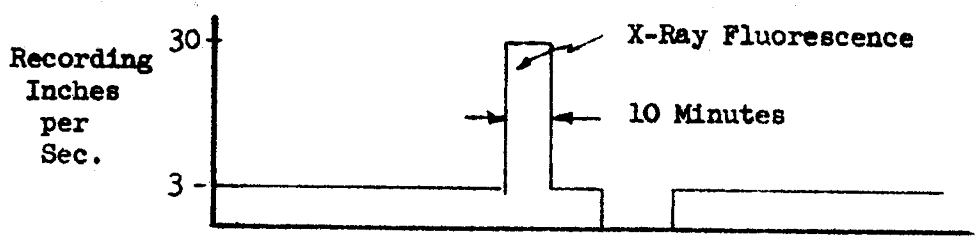


4.5.1 Communications Subsystem (Cont.)

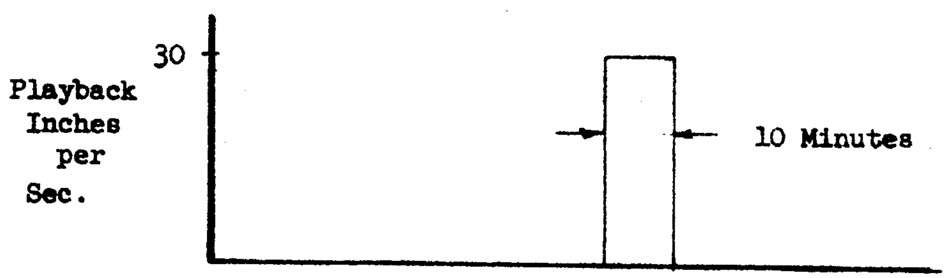
The data storage system for Case IIC is similar to that of Case IB and is shown below:



The tape recorder would be required to record at 3 ips for the low speed data and at 30 ips at the high speed X-Ray measurements. Playback would be at 30 ips. The record-playback tape speeds are illustrated in the following figures



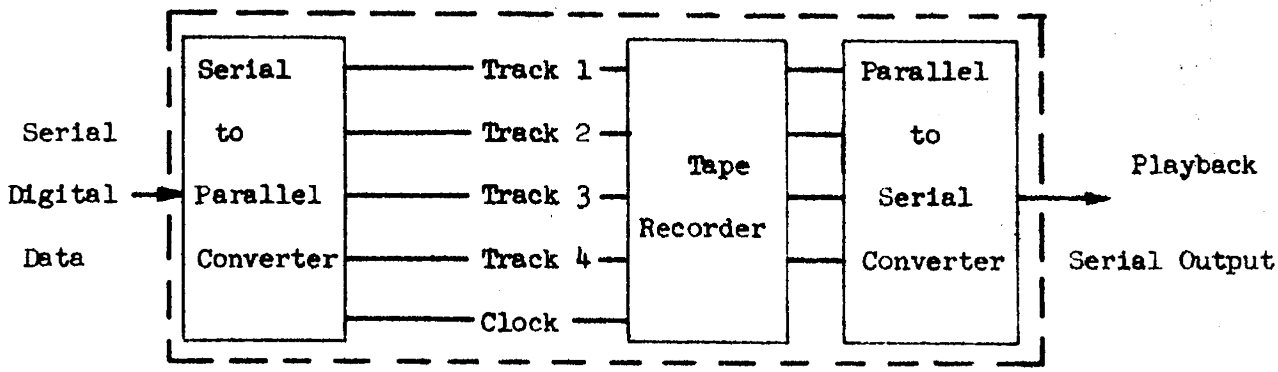
Orbit Time



Playback Speed

4.5.1 (continued)

The recorder will contain its own multiplex logic which will convert the serial (54,000 bps) to four parallel channels for storage on separate tracks on the tape. On playback the recorder reconverts the parallel data to serial data for transmission to earth, via. the high speed data link as shown in the following figure:



The data storage system, described above and applicable to all configurations, would have the following approximate characteristics:

Volume	7.1" x 7.5" x 5.7"
Weight	10 lbs.
Input Power	9 watts D.C.
Tape Speed	1, 3 and 30 inches/sec.
Tape Length	2250 feet
Tape Width	1/4 inch
Packing Density	1000 bits./inch
Multiplexing capability	Yes
Space Qualified	Yes

The current Lunar Orbiter communication subsystem must be modified only to the extent of providing a command and switching function energizing output from the tape recorder to modulation selector, in addition to the command functions controlling recording speed, on a time share basis with the video transmission function.

4.5.1 (continued)

The above modification is shown schematically in Figure 4.5.1.1. The capability provided by this modification should cover the spectrum of given experiment combinations without a significant effect on video transmission capability. With the exception of additional command requirements and decoder and programmer output modifications other subsystems, as well as the remainder of the communication subsystem, will remain unaffected.

4.5.2 **POWER SUBSYSTEM**

An analysis of the power subsystem requirements for the two mission event sequences defined in Section 3.2.3, indicates that the minimum array power requirement needed to support the maximum continuous spacecraft load and re-charge the battery at a rate which will ensure energy balance is 307.2 watts for case IB and 262 watts for case IIC. These compare to 300 watts for the Block I Lunar Orbiter Missions. In actual fact, the present Lunar Orbiter array does not meet the minimum output requirement of 300 watts at the maximum array temperature expected. But by increasing the normal battery charging rate above that required to maintain energy balance use can be made of the excess power available when array capability exceeds the load demand, so a lower minimum array output can be tolerated. In a similar manner, it can be argued that although the present array does not meet the minimum output requirements for Case IB above, there is more than sufficient energy available during the illuminated portion of the orbit to recharge the battery.

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COMMUNICATIONS SUBSYSTEM ADAPTATION

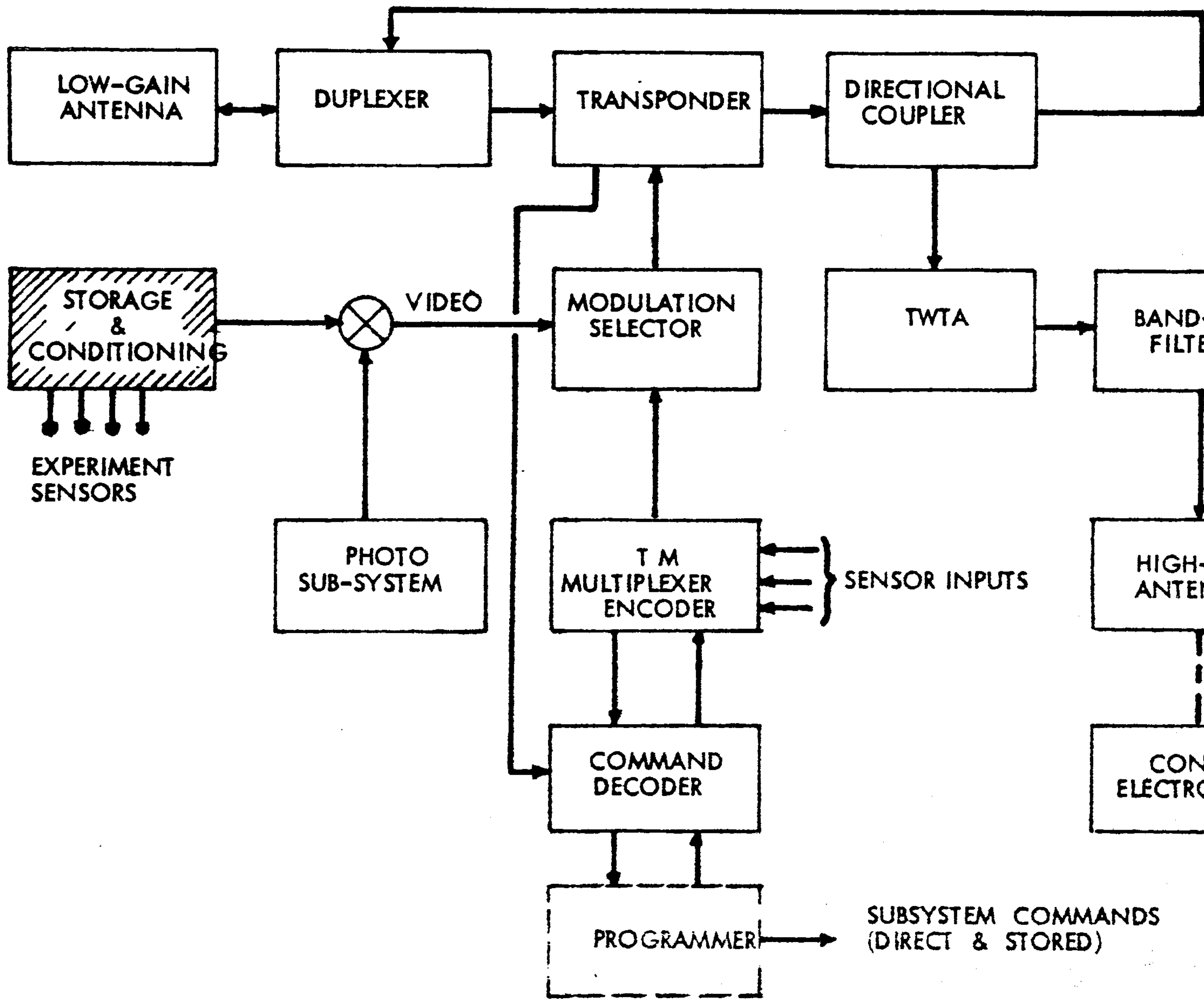

FIGURE 1.5.1.1

4.5.2 (continued)

See Figures 4.5.2.1 and 4.5.2.2. In fact, it may be seen from the following analysis that the battery charging rate required to maintain energy balance is 1.47 amps and 0.80 amps for Cases 1B and IIC respectively; thus it may not be necessary to increase the charging rate to the present limit of 2.85 $\pm$ 0.15 amps. Using a lower charging rate would reduce the risk of overheating the battery and would thus increase battery reliability.

In both cases, the battery depth of discharge in lunar orbit is not as deep as that for the Block I missions: 36.3% in Case 1B and 20.7% in Case IIC. Figure 4.5.2.5 indicates the battery depth of discharge as a function of battery capacity for both cases and the discharges with 12 and 20 ampere hour batteries, using space qualified nickel-cadmium cells. The daytime loads for the two cases are shown in Figure 4.5.2.1 through 4.5.2.4.

Summarizing, the present Lunar Orbiter Power Supply will meet the load requirements imposed by the two proposed mission event sequences, without any modification. As in the case of the Block I mission, care should be taken to see that the spacecraft loads do not exceed the array capability during the first 0.7 hours in sunlight of each orbit, or the bus voltage may fall below the minimum daylight limit. This is most likely to occur during the photo readout phase when load demands are heaviest, but with the extended daylight period of the new orbits and the judicious choice of readout times this difficulty should easily be avoided.

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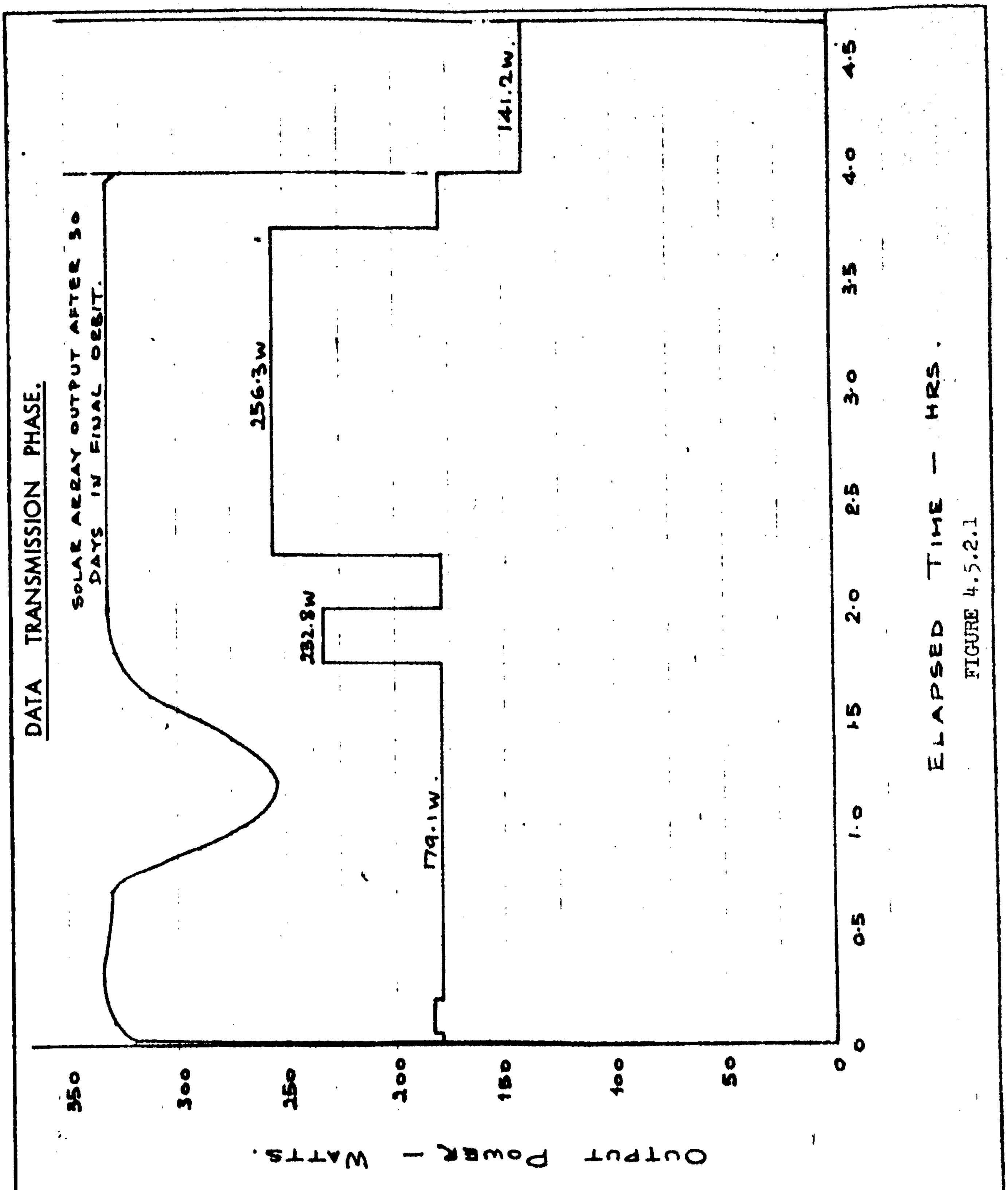

FIGURE 4.5.2.1

	INITIALS	DATE	REV BY INITIALS	DATE	TITLE	MODEL
CALC	J.G.G.	7/16/65			SUBSYSTEM LOAD PROFILE. LUNAR ORBITS. (CASE I B)	
CHECK						
APPD						
APPD						

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REV LTR________

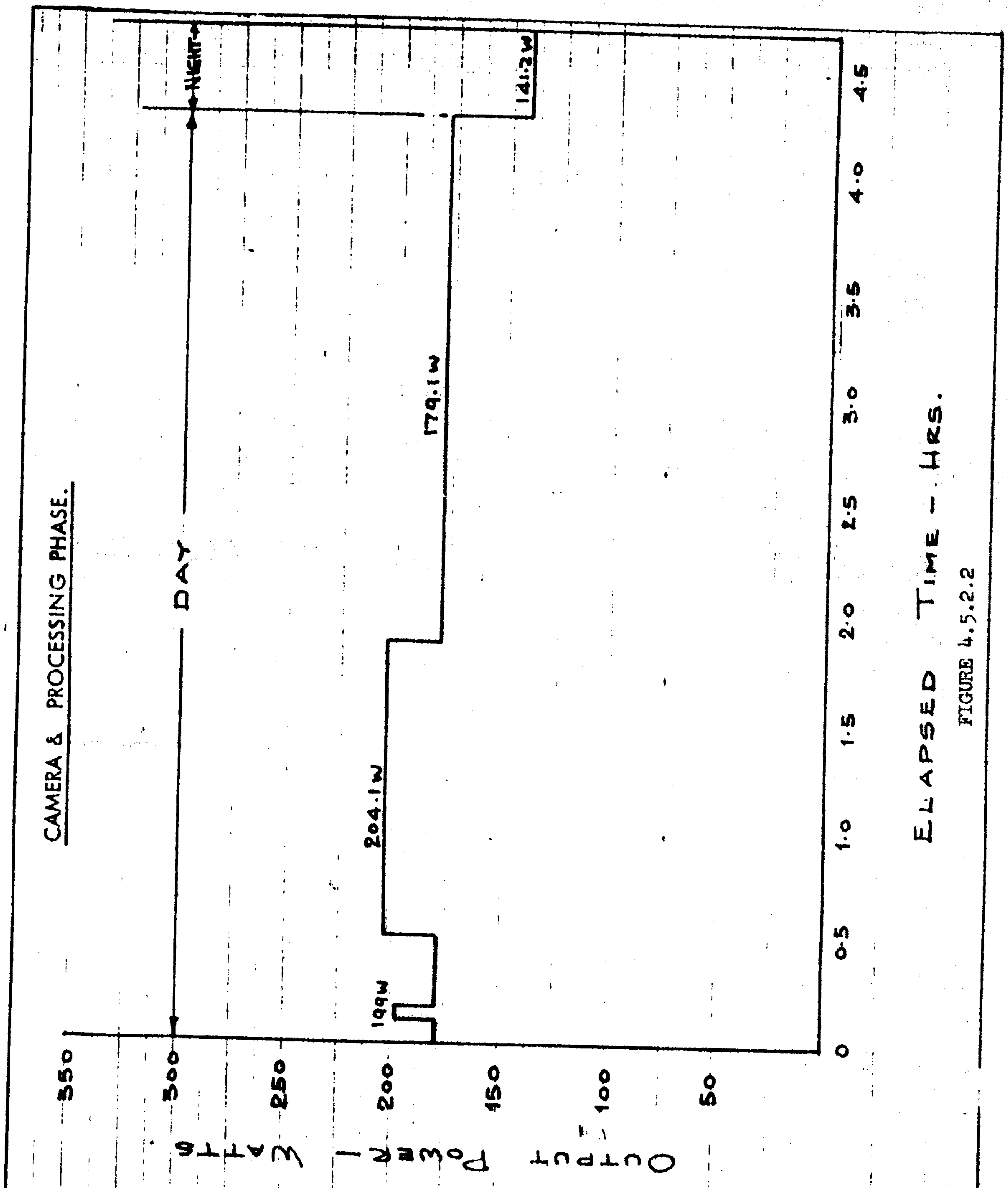

FIGURE 4.5.2.2

	INITIALS	DATE	REV BY INITIALS	DATE	TITLE	MODEL
CALC	J.G.G	7/20/65			SUBSYSTEM LOAD PROFILE	
CHECK						
APPD.					LUNAR ORBITS (CASE I B)	
APPD						

U3 4013 8000 REV. 12-64

REV LTR________

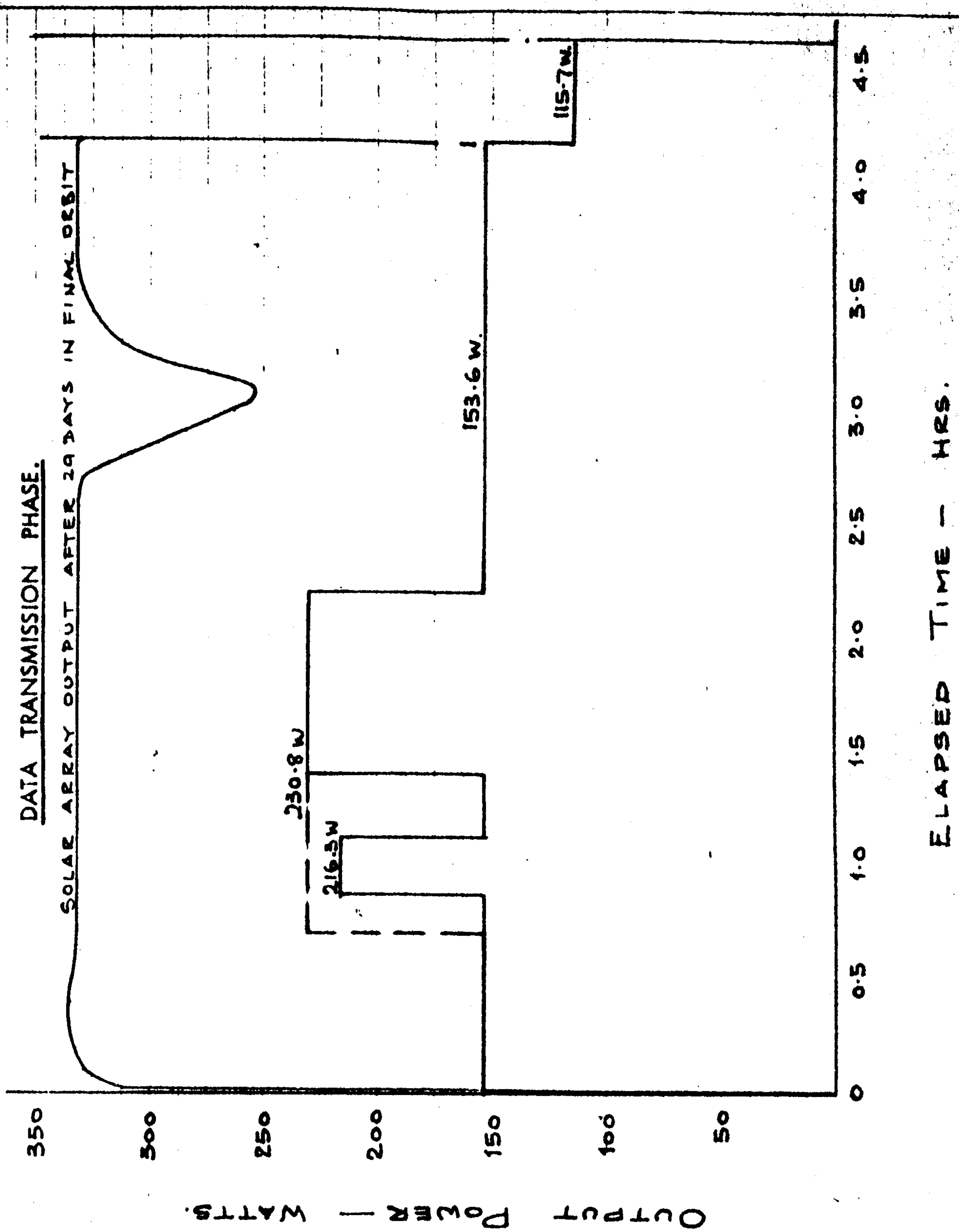

FIGURE 4.5.2.3

	INITIALS	DATE	REV BY INITIALS	DATE	TITLE	MODEL
CALC	J.G.G	7/19/65			SUBSYSTEM LOAD PROFILE LUNAR ORBITS (CASE II C)	
CHECK						
APPD						
APPD.						

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REV LTR________

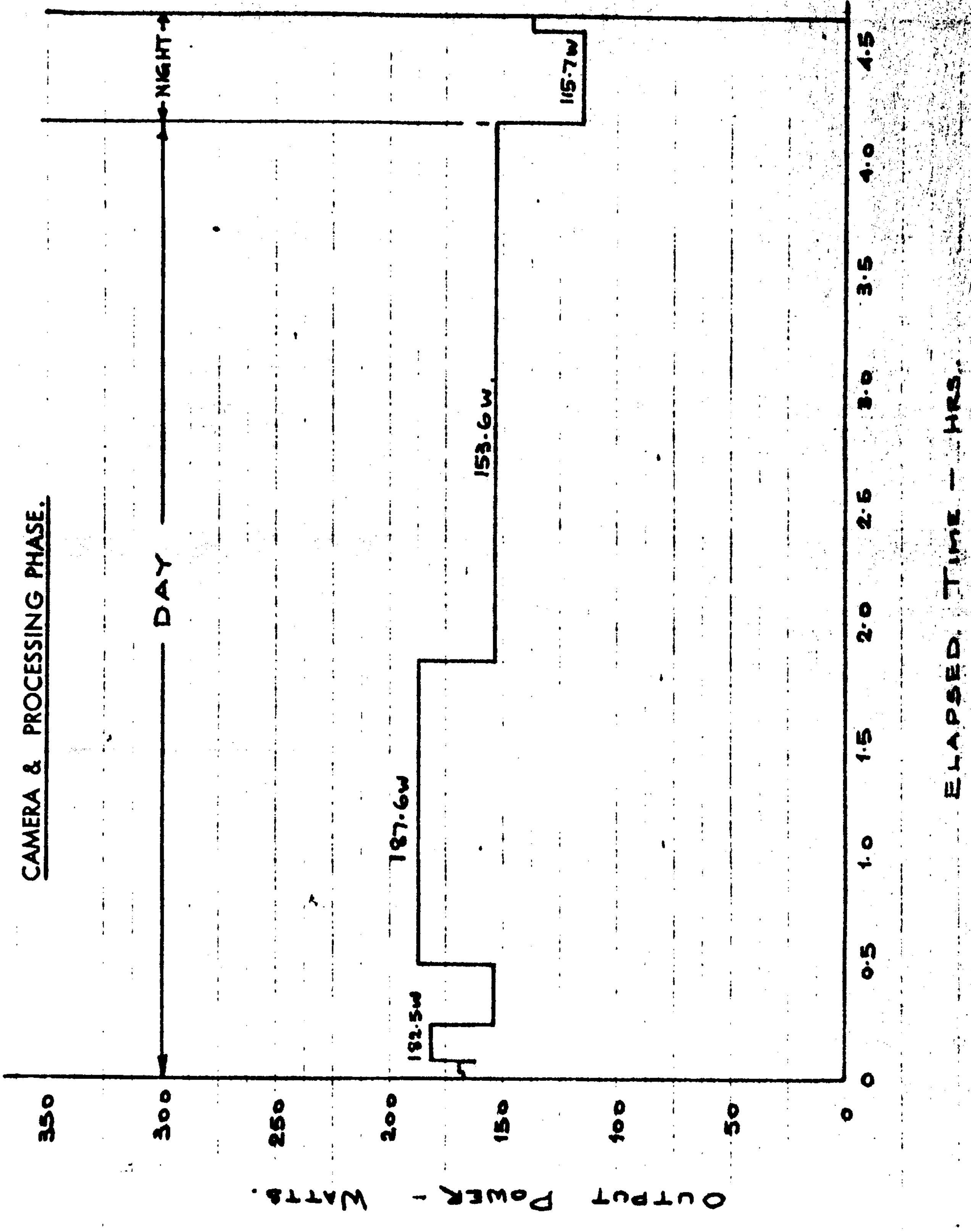

FIGURE 4.5.2.4

	INITIALS	DATE	REV BY INITIALS	DATE	TITLE	MODEL
CALC	J.G.G	7/19/65			SUBSYSTEM LOAD PROFILE LUNAR ORBITS (CASE II C)	
CHECK						
APPD.						
APPD.						

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REV LTR__________

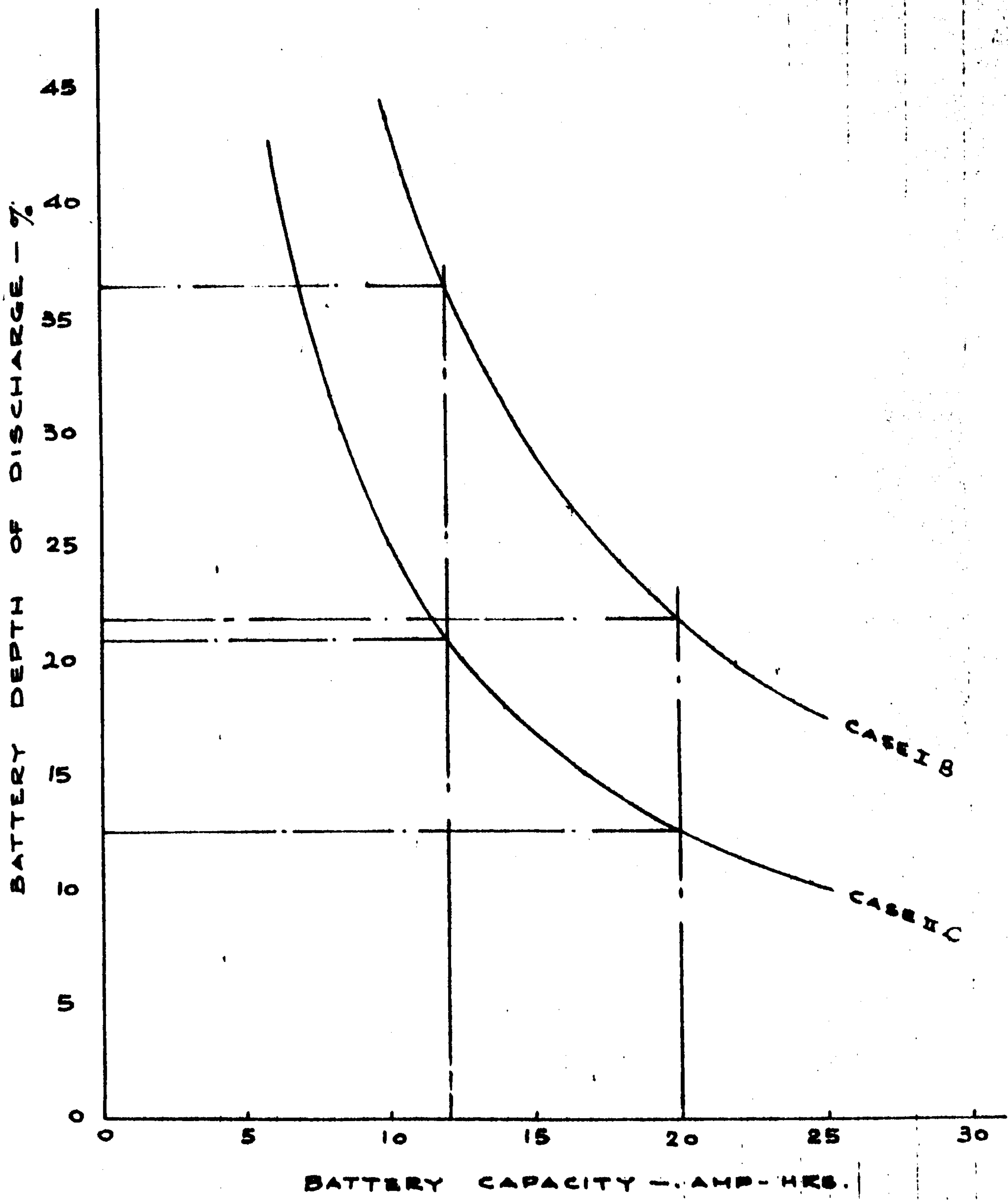

FIGURE 4.5.2.5

	INITIALS	DATE	REV BY INITIALS	DATE	TITLE	MODEL
CALC	J.G.G	7/16/65			BATTERY DEPTH OF DISCHARGE VERSUS BATTERY CAPACITY.	
CHECK						
APPD						
APPD						

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REV LTR__________

4.5.2 (continued)

Power Subsystem Analysis

Case IB

Orbital Parameters:

Apolune Altitude	3000 Km
Perilune Altitude	98 Km
Inclination	33^{o}
Illumination at Perilune	60^{o}
Orbital Period	4.69 Hrs.
Maximum Dark Time	40.6 Mins.
	= 0.68 Hrs.

Analysis

List of Symbols:

I_a	-Solar Array Current
I_{bd}	-Battery Node Current
I_{bc}	-Battery Charging Current
I_{RL}	-Current Losses in Charge Controller and Shunt Regulator
I_S	-Shunt Regulator Current
I_c	-Charge Controller Current
I_L	-Load Current at Power Output Connector
V_a	-Array Voltage
V_b	-Battery Voltage
V_d	-Drop Across Blocking Diode D_1
P_{oq}	-Power Output
e_b	-Battery Charge - Discharge Efficiency at a Given Temperature
T	-Orbital Period
t_n	-The Night Time Portion of the Orbit
t	-Time after Start of the Nighttime Portion of the Orbit

From the load summary of section 3.2.3, it is seen that the basic load requirements are 106.5 watts in the daytime and 100 watts at night. The battery discharge current at night is given by:

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4.5.2 (continued)

$$I_{bd} = I_{RL} + I_s + I_L$$

$$= 0.267 + 0 + \frac{P_o}{V_b - V_d}$$

$$= 0.267 + 0 + \frac{99.7}{24 - 1} = \underline{\underline{4.60 \text{ amps}}}.$$

Thus the basic night-load before we add instrumentation for experiments is 4.60 amps.

We may now add the instrumentation and photography power requirements.

<u>Instrumentation</u>

	Power Required Watts Day	Power Required Watts Night
Full Photo Capability HR 1 meter LR 9 meter, Readout Mode	69.8	15.0
Micro-Meteoroid Experiment	1.5	1.5
Solar Plasma Experiment	8.0	8.0
Magnetometer Experiment	7.0	7.0
Photometry/Colorimetry Experiment	4.0*	--
Subtotal	86.3	31.5

*Does not occur concurrently with readout.

4.5.2 (continued)

Communications

The TWT amplifier and high gain antenna will be required.	Power Required Watts Day	Night
High Gain Antenna Controller	0.5	0.5
TWT Amplifier	54.0	0.5
Tape Recorder	9.0	9.0
Subtotal	63.5	10.0
TOTAL	149.3 W	41.5 W

Total Battery Discharge at Night

$$I_{bd} = 4.60 + \frac{41.5\ W}{23\ V}$$

$$= \underline{6.404\ \text{amps}}$$

If we assume that during orbital daytime $I_{bd} = 0$, then equation for energy balance is:

$$\frac{I}{eb} \int_{t_n}^{T} I_c dt = \int_{t=0}^{t_n} I_{bd} dt \quad \ldots\ldots (1)$$

Substituting values in equation (1) gives

$$\frac{I}{e_b} \int_{0.68}^{4.69} l_c dt = \int_{t=0}^{t=0.68} 6.404\ dt$$

4.5.2 (continued)

Integrating the above expression gives

$$\frac{I_c}{e_b} (4.69 - 0.68) = 4.355 \text{ amp hours}$$

In the worst case $e_b = 1.35$ at a battery temperature of 35°C

Thus $I_c = \dfrac{4.355 \times 1.35}{4.01}$

$= \underline{\underline{1.47 \text{ amps}}}$

This is the minimum charging current for energy balance in the worst case.

Minimum Solar Array Requirements

The array current required to supply the maximum continuous daytime loads, charge the battery and supply subsystem losses is:

$$I_a = I_L + I_c + I_s + I_{RL} \qquad \ldots\ldots (2)$$

In the "off" mode $I_s = 0$ amps

$I_c = 1.47$ amps

$I_{RL} = 0.267$ amps

and $I_L = \dfrac{106.5 + 149.8}{V_a} = \dfrac{256.3}{29.3} = 8.747\text{A}$

NOTE: $V_a = V_b + V_c$

$V_b = 20 \times 1.44$ volts/cell (at 35°C) = 28.8V

$V_c = 0.5$ volts

$V_a = 28.8 + 0.5 = \underline{29.3 \text{ Volts}}$

4.5.2 (continued)

Substituting the above values in equation (2) gives

$$I_a = 8.747 + 1.47 + 0 + 0.267$$

$$= \underline{10.484 \text{ amps}}$$

The minimum required power from the array

$$P_a = I_a V_a = 10.484 \times 29.3V$$

$$= \underline{307.2 \text{ watts}}$$

Battery Depth of Discharge

The battery discharge current at night was shown to be 6.404 amps. With a maximum dark period of 0.68 hours, the battery discharge is 4.355 amp. hours, or a 36.3% depth of discharge for the 12 amp-hour battery.

Case IIC

Orbital Parameters:

Apolune Altitude	3000 Km
Perilune Altitude	46 Km
Inclination	45^o
Illumination at Perilune	90^o
Orbital Period	4.64 Hrs.
Maximum Dark Time	27.44 Mins.
	= 0.44 Hrs.

Analysis

Power Requirements: The basic load requirements are 106.5 watts in the daytime and 99.7 watts at night.

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4.5.2 (continued)

For Case IIC, the instrumentation and photography power requirements are as follows:

Instrumentation

	Power Required Watts Day	Power Required Watts Night
Full photo capability HR 1 meter LR 8 meter readout mode	69.8	15.0
Radio meter experiment	5.0*	5.0*
Infrared experiment	--	8.0*
X-Ray Fluorescence experiment	2.0**	--
Photometry/Colorimetry experiment	4.0**	--
Subtotal	69.8	15.0

* The Radiometry experiment will be initiated approximately 4 minutes prior to crossing the terminator and will be terminated approximately 4 minutes after crossing the terminator. The IR experiment will be initiated approximately 4 minutes prior to crossing the terminator and will end at the terminator.

** Does not occur concurrently with readout.

Communications

The TWT amplifier and high gain antenna will be required.

	Power Required Watts Day	Power Required Watts Night
High Gain Antenna Controller	0.5	0.5
TWT Amplifier	54.0	0.5
Subtotal	54.5	1.0
TOTAL	124.3	16.0

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4.5.2 (continued)

From Case IB it is seen that the basic nighttime load current is 4.60 amps.

Total continuous battery discharge at night.

$$I_{bd} = 4.60 + \frac{16W}{23V}$$

$$= \underline{5.3 \text{ amps.}}$$

The equation for energy balance, assuming $I_{bd} = 0$ during orbital daytime, is:

$$\frac{I}{e_b} \int_{t_n}^{T} I_c dt = \int_{0}^{t_n} I_{bd} dt + \left(\frac{13W}{23V} = 0.07 \text{ hrs}\right) \quad \ldots\ldots\ldots(1)$$

The last factor on the R.H. side is the discharge attributable to the IR and Radiometry experiments during orbital nighttime.

Again, in the worst case $e_b = 1.35$ at a battery temperature of 35^oC.

Substituting values in equation (1)

$$\frac{I}{1.35} \int_{0.46}^{4.64} I_c \, dt = \int_{0}^{0.46} 5.3 \, dt + 0.04$$

$$\frac{I_c}{1.35} (4.64 - 0.46) = 2.44 + 0.04 \text{ amp hours}$$

$$I_c = \frac{2.48 \times 1.35}{4.18} \text{ amps}$$

$$= \underline{0.80 \text{ amps}}$$

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4.5.2 (continued)

Minimum Solar Array Requirements

$$I_a = I_L + I_c + I_s + I_{RL}$$

$$= \frac{106.5 + 124.3W}{29.3V} + 0.8 + 0 + 0.267$$

$$= 7.877 + 0.8 + 0.267$$

$$= \underline{8.944 \text{ amps}}$$

The minimum required power from the array

$$P_a = I_a V_a = 8.944 \times 29.3V$$

$$= \underline{262 \text{ Watts}}$$

Battery Depth of Discharge

The battery discharge at night was 2.48 amp-hours, or 20.7% depth of discharge for the 12 amp-hour battery.

4.5.3 THERMAL CONTROL

The Case IB configuration will provide the same component thermal environment as the existing Lunar Orbiter design with the exception of the command decoder. The command decoder would be subjected to a minimum temperature of -15°F instead of its present minimum of +10°F.

Because of the increased apolune and perilune altitudes, increased inclination and location of orbit perilune the mission design will subject the equipment mounting check (EMD) continuously to exposure to sunlight for the first six days in lunar orbit. The present design of the spacecraft thermal control limits the power loads to

4.5.3 (continued)

110 watts under the above conditions. A modification in the EMD exterior paint would improve the spacecraft capability with respect to maximum power load capability under 100% sunlight conditions. As can be seen by reference to Figure 4.4.0.1b the photographic sequence, with an associated power output of 179 watts, is scheduled to commence at 3 days after orbit injection. Photography and film processing, requiring a total power load of 204 watts would be carried out prior to the time when spacecraft sun occultation occurs. Solar energy input to the EMD would be reduced during the photography sequence by approximately 15% due to the solar incidence 33^{o} with respect to the deck. This attitude can be maintained as long as necessary during the remainder of the orbit which will effectively improve the spacecraft thermal capability. Furthermore, if the above solar energy reduction in conjunction with an EMD exterior point change does not provide sufficient capability a deliverate maneuver, orienting the EMD away from the sun for a period sufficient to restore thermal balance can be executed at some penalty in attitude control gas.

As the Case IB mission progresses the sun occultation time(s) approaches the current design limit of 80% of orbital time in the light under a maximum power load of 250 watts. Additionally. as can be seen by reference to the power subsystem analysis in section 4.5.2, a constant load of 250 W is not contemplated.

4.5.3 (continued)

The Case IIC mission results in an average of 91% of orbital time in the sun. This is 11% in excess of the maximum current design time of 80% under a maximum power load of 250 W. It should be noted that, by reference to the power load analysis, that the maximum power load of 250 W is limited to 16% of orbital time prior to completion of the 30 day reconnaissance mission and 32% during the retransmission of film data after 30 days. Furthermore, during a period of 9% of the orbit the solar energy input is reduced by 29% due to misalignment of 45° from the sun due to experiment orientation requirements. The above misorientation can be maintained for a longer time, as required for restoration of thermal balance, without penalty in attitude control gas.

In summary, both of the above configurations and missions indicate a need for improvement of thermal capability of the spacecraft for high orbit inclinations. This improvement can be achieved by either exterior paint change or deliberate misalignment from sun orientation or both if necessary. Detailed computer analysis will be required to determine the extent of modifications required. This should be carried out when more precise experiment definitions, including their power dissipation, thermal control and solar shielding requirements, become available.

4.5.4 VELOCITY CONTROL

The parametric trajectory design data of Section 3.2.1 used in the definition of mission profiles for Cases IB and IIC was verified by detailed trajectory computations. The data of figures 4.5.4.1 and 4.5.4.2 present the velocity requirements as a function of longitude of arrival for May and July of 1967. The superimposed horizontal lines represent the available velocity increment after budgeting of 117 m/sec. for midcourse error correction, injection error correction, specific impulse degradation, finite burn time and propellant off-loading. It is to be noted that three launch days per month are available for each one of the missions. This should in general constitute an operationally acceptable launch period. The abscissa of Figures 4.5.4.1 and 4.5.4.2 represents a measure of waiting time in orbit when an experiment is to be performed at a specific longitude in the near equatorial region. The conversion can be achieved by differencing the arrival longitude and target longitude and dividing by the rate of rotation of the moon (12.6°/day).

In the case of the Case IIC mission the mapping mission would be initiated at an approximately 80° East longitude.

4.5.5 ATTITUDE CONTROL

An analysis of the mission profiles shown in Section 3.2.2 was carried out in order to determine the extent of attitude control subsystem modification requirements. The two mission-configuration profiles pose a problem of providing sufficient attitude control

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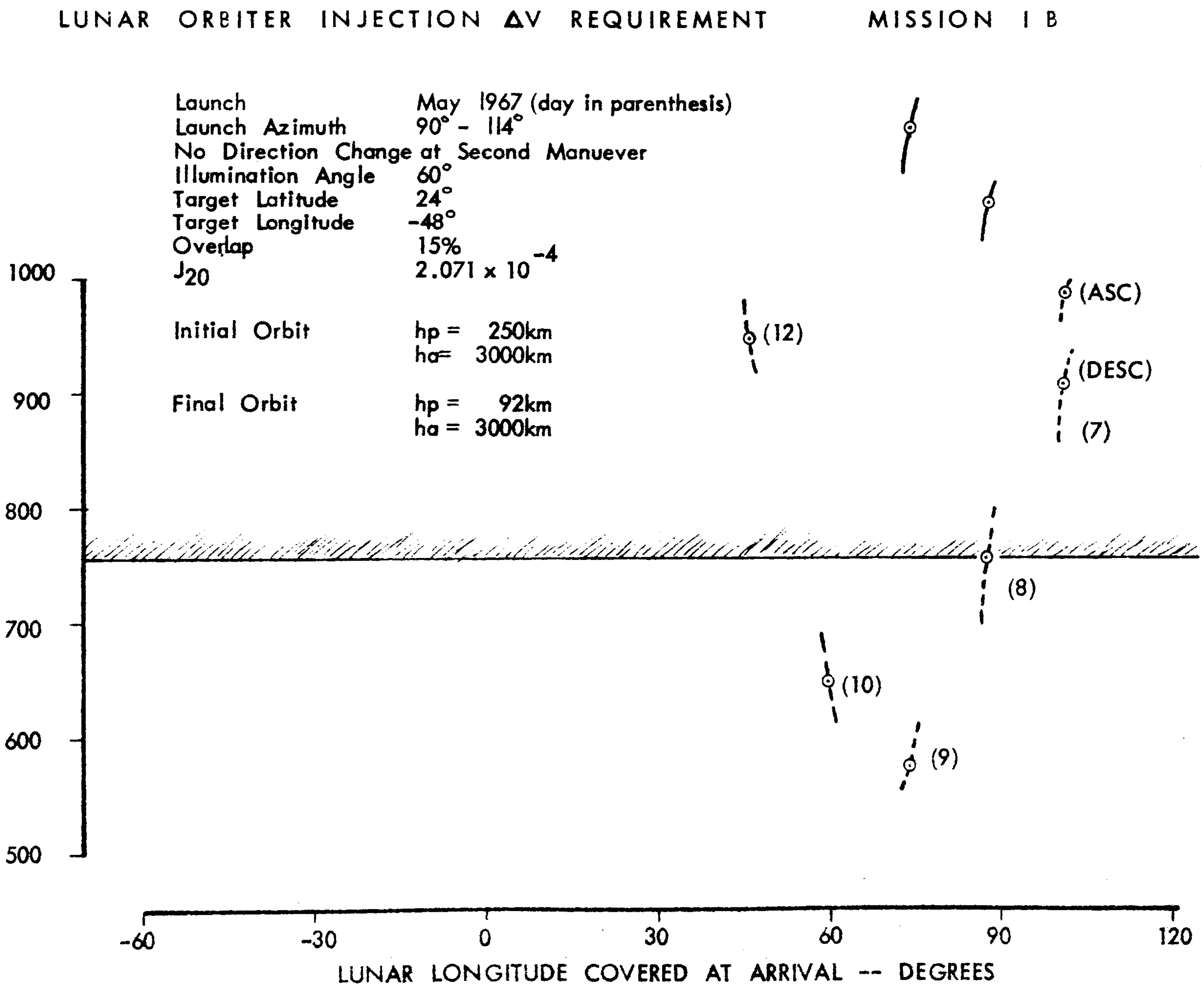

FIGURE 4.5.4.1

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LUNAR ORBIT INJECTION ΔV REQUIREMENTS

L/O SCIENTIFIC MISSION II C

INITIAL ORBIT hp = 250 ha = 3000 | FINAL ORBIT hp = 46 ha = 3000 | ILLUMINATION ANGLE = 90° (3 - DAY WAIT) | TARGET LATITUDE = 26° (PERILUNE)

NO DIRECTION CHANGE AT 2nd MANEUVER | 43° INCLINATION | $J_{20} = 2.071 \times 10^{-9}$

90° - 114° LAUNCH AZIMUTH

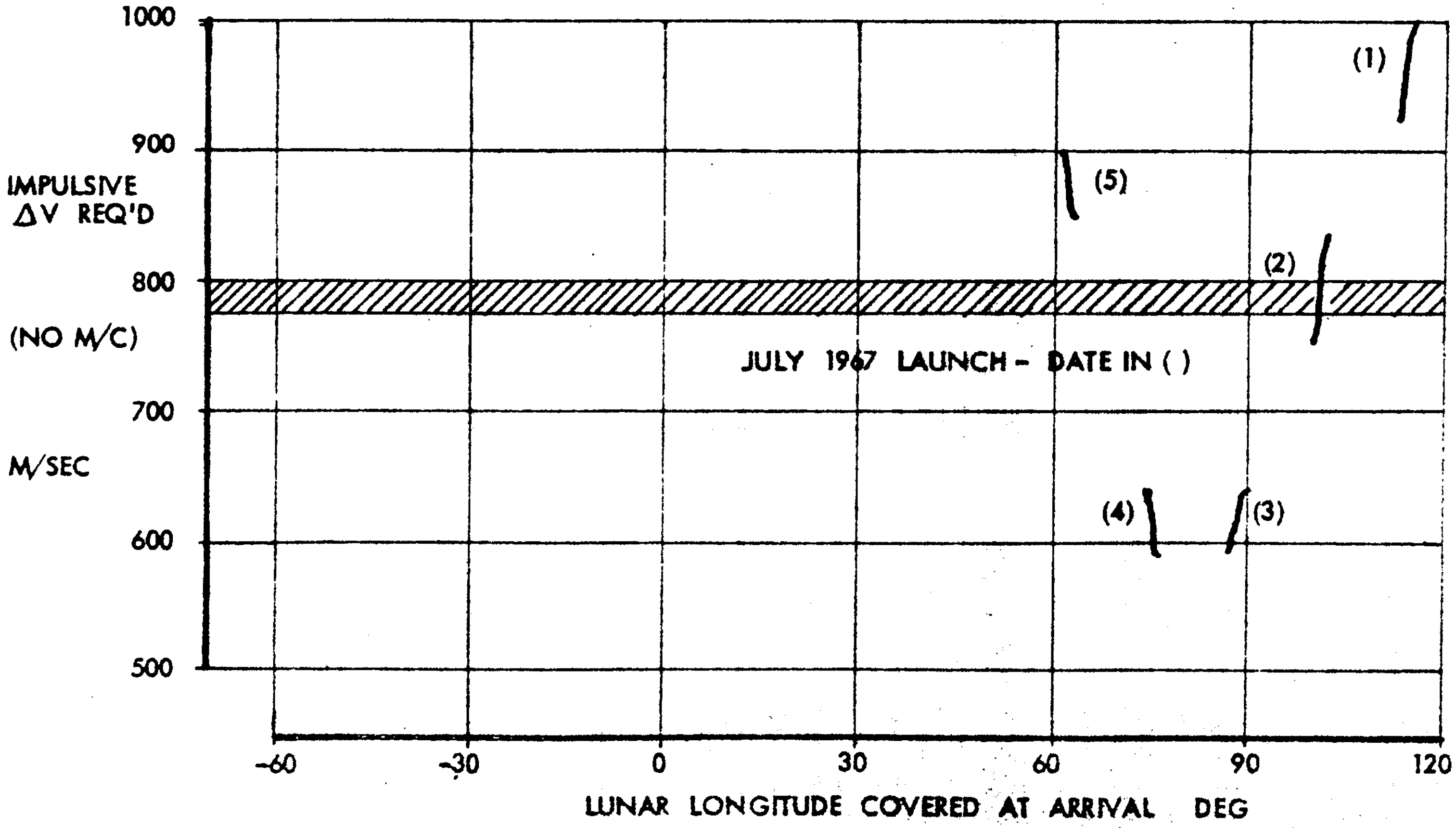


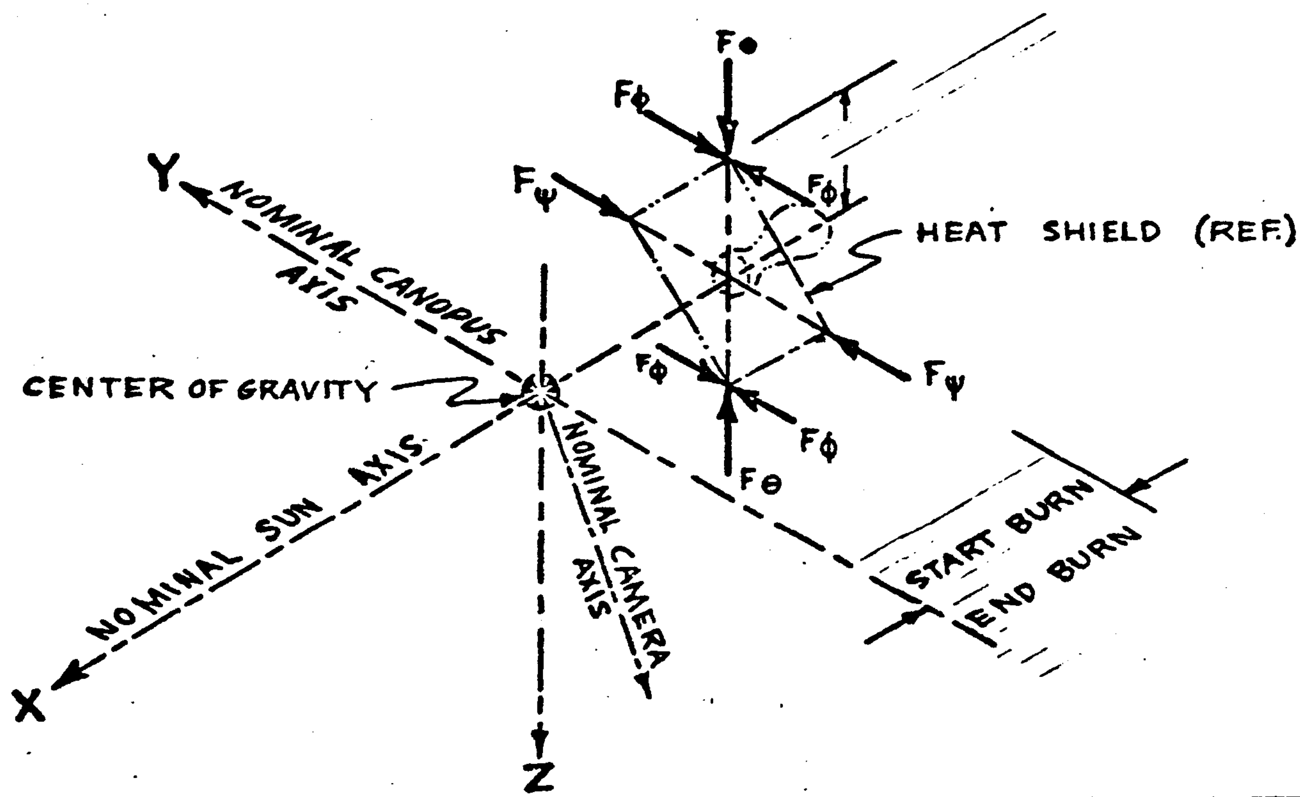
FIGURE 4.5.4.2

4.5.5 (continued)

gas budget for an extended life mission which can be solved by several alternate means.

The Case IB configuration represents an average increase of moments of inertia of 2.6 relative to the present spacecraft configuration. The moments of inertia for this case are shown in Figure 4.5.5.1. If the current spacecraft maneuver rates were to be preserved a proportionately increased nitrogen gas flow rate would be required. The increased flow rate would require thruster size and regulator modification with the possible additional modification of supply line due to pressure drop under increased flow rate. These modifications would require investigation in greater detail. Assuming the above modifications, an increased nitrogen gas capacity would be required for the Case IB configuration and its associated mission. This is shown by the tabulation of Figure 4.5.5.2 which states a requirement for 24.78 pounds of gas, relative to the presently available 10 pounds, for the specified mission followed by an extended mission life-time of one year. The above nitrogen gas increment, of 14.78 pounds, would require a spacecraft weight increment of 38.5 pounds because the tankage weight can be estim ated at 1.6 times the nitrogen weight. Weight, increments associated with extended lifetime requirements of less than one year can be estimated by reference to Figure 4.5.5.3.

L.O. ADAPTIBILITY STUDY
PERFORMANCE DATA



CASE II		INERTIA SLUG FT^2	REACTION CONTROL Σ JET FORCE	LEVER ARM	ACCELERATION	THRUST VECTOR CONTROL C.G TO GIMBAL	Accel. °/S^2/°
START BURN	PITCH	260.58	.075	32.24	.052	32.24	1.03
	ROLL	301.75	.075	17.14	.049		
	YAW	217.78	.06	32.24	.051	32.24	1.24
END BURN	PITCH	214.27	.075	42.65	.085	42.65	2.0
	ROLL	292.14	.075	17.14	.05		
	YAW	168.28	.06	42.65	.057	42.65	2.54

FIGURE 4.5.5.1

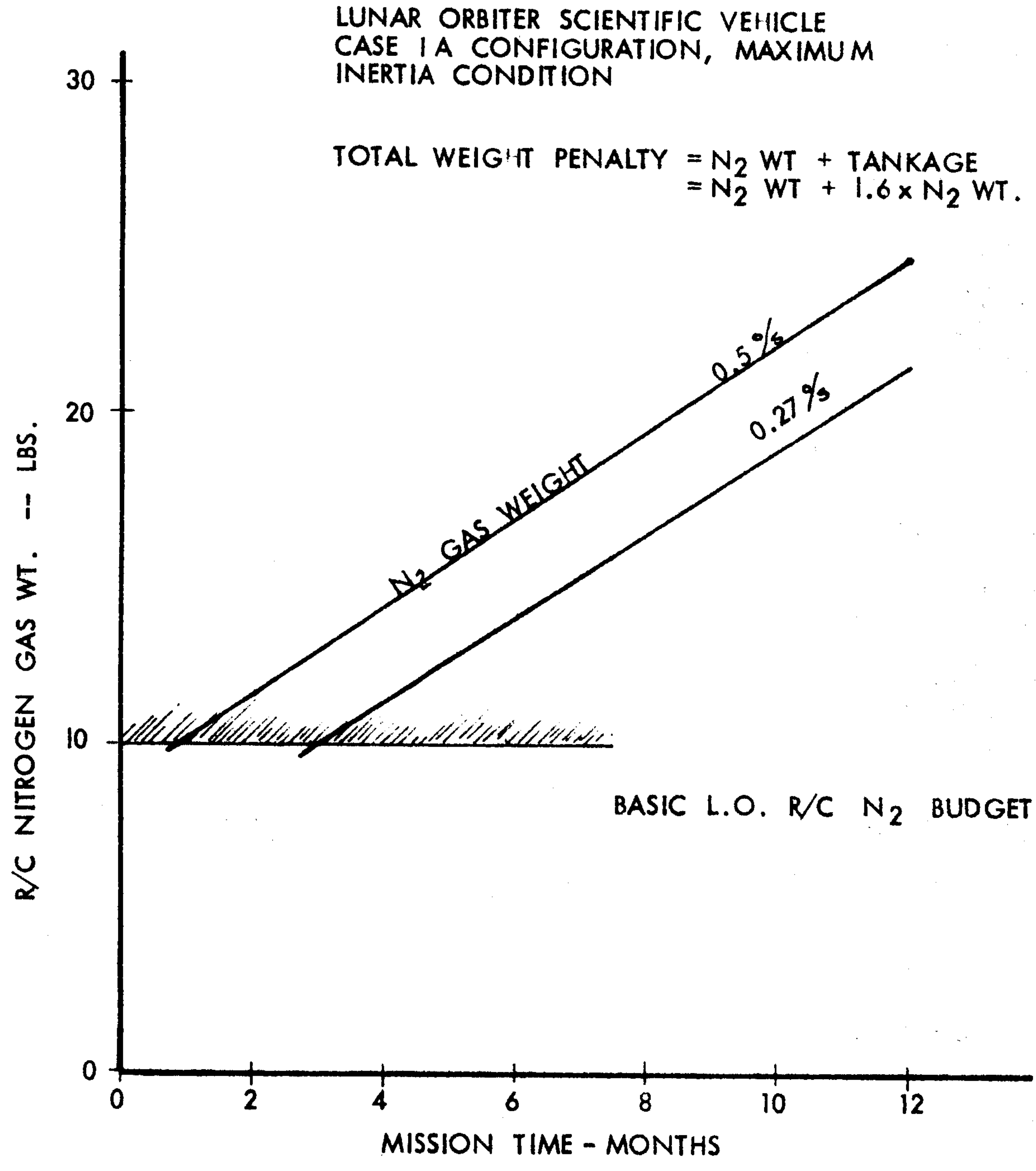
FIGURE 1

CASE IB MISSION

REACTION CONTROL NITROGEN WEIGHT BUDGET

MISSION PHASE	NITROGEN WEIGHT - LBS			
	BASIC VEHICLE		SCIENTIFIC VEHICLE	
	ITEM	TOTAL	ITEM	TOTAL
PHOTOGRAPHIC MISSION				
(1) Initial Acquisition	.22		.57	
(2) Translunar Coast	.06		.16	
(3) Midcourse Maneuvers				
(a) 1st Midcourse	.14		.36	
(b) 2nd Midcourse	.18		.47	
(4) Initial Orbit Injection	.14		.36	
(5) Final Orbit XFR	.09		.23	
(6) Photo Maneuvers (12)	1.19		3.10	
(7) Photo Transmission (10 days)	.68		1.80	
(8) Lunar Orbit Coast (17 days)	.12		.31	
(9) Celestial Reacquisition	.84		2.20	
(10) Disturbances	.14		.70	
(11) R/C Cross Coupling	.20		.52	
Photo Mission Total		4.00		10.78
EXTENDED MISSION				
(1) Lunar Orbit Coast	1.79		4.65	
(2) Celestial Reacquisition	.56		1.45	
(3) Disturbances	1.48		7.40	
(4) R/C Cross Coupling	.19		.50	
Extended Mission Total		4.02		14.00
Reserve		1.98		0
TOTAL NITROGEN BUDGET		10.00		24.78

FIGURE 4.5.5.2



REACTION CONTROL FUEL REQUIREMENT
LUNAR ORBITER SCIENTIFIC VEHICLE -- CASE 1A

FIGURE 4.5.5.3

CASE IB MISSION

REACTION CONTROL NITROGEN WEIGHT BUDGET

MISSION PHASE	NITROGEN WEIGHT - LBS			
	BASIC VEHICLE		SCIENTIFIC VEHICLE	
	ITEM	TOTAL	ITEM	TOTAL
PHOTOGRAPHIC MISSION			1	
(1) Initial Acquisition	.22		.57	
(2) Translunar Coast	.06		.16	
(3) Midcourse Maneuvers				
(a) 1st Midcourse	.14		.18	
(b) 2nd Midcourse	.18		.24	
(4) Initial Orbit Injection	.14		.18	
(5) Final Orbit XFR	.09		.12	
(6) Photo Maneuvers (12)	1.19		1.55	
(7) Photo Transmission (10 days)	.68		1.80	
(8) Lunar Orbit Coast (17 days)	.12		.31	
(9) Celestial Reacquisition	.84		1.10	
(10) Disturbances	.14		.70	
(11) R/C Cross Coupling	.20		.52	
Photo Mission Total		4.00		7.43
EXTENDED MISSION				
(1) Lunar Orbit Coast	1.79		4.65	
(2) Celestial Reacquisition	.56		1.45	
(3) Disturbances	1.48		7.40	
(4) R/C Cross Coupling	.19		.50	
Extended Mission Total		4.02		14.00
Reserve		1.98		0
TOTAL NITROGEN BUDGET		10.00		21.43

1 Maneuver Rate = .27 deg/sec = (half nominal)

Maneuver Fuel = 1.3 x basic (inertia + rate effect)

Other Fuel = 2.6 x basic (inertia effect)

Extend Mission Fuel = $\frac{14.00}{335}$ = .042 lb/day

Extend Mission = $\frac{(10.00 - 7.43)}{.042}$ = 61 days

Total Mission = 91 days

FIGURE 4.5.5.4

USE FOR TYPEWRITTEN MATERIAL ONLY

4.5.5 (continued)

In any case, if maneuver rates are preserved for this configuration, extended lifetime capability would have to be attained by a one-to-one exchange of experimental payload for additional attitude control gas and tankage. The exchange rate is approximately 3.5 pounds of experiment payload per month of extended lifetime in excess of 1 month.

A preferable alternate approach to the problem of the increased moments of inertia is to accept reduced maneuver rates. This would eliminate the necessity for the control system modifications discussed above but would require a change in the closed loop electronics to accommodate the decreased rate. The attitude control gas budget would be greatly improved by this modification. For example, a reduction of the maneuver rates by a factor of two would result in a 3-month extended life capability with a total budget of 10 pounds of nitrogen. An attitude gas budget under these ground rules is shown in Figure 4.5.5.4 and other budget distributions can be obtained by reference to Figure 4.5.5.3. A reduction of the rates by an average factor of 2.6 would yield the same lifetime capability as the present configuration.

In order to capitalize on this possibility the mission design would have to insure adequate time for performing the attitude maneuver after the spacecraft emerges into sunlight if the experiment orientation accuracy is critical. In the case of the IB mission this does not present any problem inasmuch as the spacecraft is 100% in the sunlight during the days when

4.5.5 (continued)

surface related experiments were executed. Generally it would appear that times of 30-40 minutes can be achieved, allowing a reduction of rates by at least a factor 2:1 relative to the current design, with some care in mission design. This solution to the problem of attitude control gas budget appears, therefore, to be very attractive for configurations with high moments of inertia due to boom deployment and increased weight.

The Case IIC configuration preserves the moments of inertia of the standard configuration. The problem in this case is the increased maneuver gas consumption due to the increased number of maneuvers stipulated in this general reconnaissance mission. The maneuver gas budgeting for this mission is summarized in Figure 4.5.5.5. It is to be noted that a budget of only .55 pounds remains for extended lifetime. An additional 5.45 pounds of nitrogen would be required to provide for an extended lifetime of one year and a reserve of 2 pounds. This can be assured by additional allowance of 14.3 pounds of nitrogen and tankage which is feasible in the Case IIC configuration since a 21 pound total weight margin was found to be available in this case. The requirement for additional nitrogen would be achieved by addition of manifolded tanks rather than an increase in the size of the present tank because of volume limitations at the present tank location.

CASE IIC MISSION

REACTION CONTROL NITROGEN WEIGHT BUDGET

MISSION PHASE	N_2 WT. LBS BASIC VEHICLE ITEM	N_2 WT. LBS BASIC VEHICLE TOTAL
PHOTOGRAPHIC MISSION		
(1) Initial Acquisition	.22	
(2) Translunar Const	.06	
(3) Midcourse Maneuvers		
(a) 1st Midcourse	.14	
(b) 2nd Midcourse	.18	
(4) Initial Orbit Injection	.14	
(5) Final Orbit XFR	.09	
(6) Photo Maneuvers (44)	5.28	
(7) Photo Transmission	1.89	
(8) Lunar Orbit Coast	.27	
(9) Celestial Reacquisition	.84	
(10) Disturbances	.14	
(11) R/C Cross Coupling	.20	
Photo Mission Total		9.45
Reserve		0.55
TOTAL NITROGEN BUDGET		10.00

FIGURE 4.5.5.5

APPENDIX A

LIST OF TYPICAL EXPERIMENTS
FOR FOLLOW-ON MISSIONS
ON
LUNAR ORBITER

LUNAR ORBITER SCIENTIFIC MISSION PLANNING

EXPERIMENT NO.	TYPICAL EXPERIMENTS	CIS-LUNAR	LUNAR ORBITS ELLIPTIC	LUNAR ORBITS CIRCULAR
1*	Gamma-Radiation (Isotopic Composition)	Continuous	Acceptable Intermittent	Desired Coutinuous
2*	Infrared (Thermal Mapping & Identification of Minerals)	Off	Acceptable Intermittent	Desired Continuous
3	Bi-Static Radar (Radar Scatter Cross Section, Correlation Functions, Dielectric Constant)	Off	Acceptable Intermittent	Desired Continuous
4	Micrometeoroid (Primary & Secondary Ejecta)	Continuous	Continuous	
5	Solar Plasma	Continuous	Coutinuous	
6	Magnetic Field	Continuous	Continuous	
7	(a) Photometry (Surface Features)	Off	Acceptable Intermittent	Desired Continuous
7	(b) Colorimetry (Differentiation of Material)	Off	Acceptable Intermittent	Desired Continuous
8	X-Ray Fluorescence (Elemental)	Off	Acceptable Intermittent	Desired Continuous
9	Radiometer	Off	Acceptable Intermittent	Desired Continuous
10	Selenodesy	Off	Desired Intermittent	

Inclinations

Equatorial to Polar - - - - (applies to all experiments)

* Strive for same inclinations used for Photomissions to permit coverage of same areas already photographed and mapped to allow correlation with Gamma-ray and Infrared Data.

Note: Page 2 is a continuation (going across) of Page 1.

SPECIAL REQUIREMENTS

XPERIMENT NO.	ATTITUDE OF SPACECRAFT ACCEPTABLE	ATTITUDE OF SPACECRAFT DESIRED	DATA HANDLING ACCEPTABLE	DATA HANDLING DESIRED	MOUNTING	COMMANDS	TELEMETRY
1	Part Time View of Surface	Continuous View of Surface	Real Time	Storage	Boom Req'd Look to Surface	Real Time	500 bps
2	Part Time View of Surface	Continuous	Real Time	Storage	Look to Surface	Real Time and Stored	500 bps
3	Stabilized in One or More Axes	Spin Stabilized		Real Time	Isotropic Antennas	Real Time and Stored	2 KC 20 cps, SCO 40 cps, SCO
4		Stabilized in one or more Axes	Real Time	Storage	Toward and away from Surface	Real Time and Stored	200 bpm
5	Stabilized in one or more Axes	Spin Stabilized	Real Time	Storage	Boom Required	Real Time and Stored	500 bps
6	Stabilized in one or more Axes	Spin Stabilized	Real Time	Storage	Boom Required	Real Time and Stored	100 bps
7	Part Time View of Surface	Continuous View of Surface	Real Time	Storage	Look to Surface	Real Time and Stored	500 bps
7	Part Time View of Surface	Continuous View of Surface	Real Time	Storage	Look to Surface	Real Time and Stored	500 bps
8	Part Time View of Surface	Continuous View of Surface	Real Time	Storage	Look to Surface	Real Time and Stored	54,000 bps
9	Part Time View of Surface	Continuous View of Surface	Real Time	Real Time	Isotropic Antennas	Real Time and Storage	500 bps
10	Loose Control $\pm 12^{0}$ attitude limit cycle			Real Time		None Req'd	Transponder operative

APPENDIX B

LIST OF EXPERIMENTS WITH

DESCRIPTIONS

1. GAMMA RADIATION EXPERIMENT

Objectives: Determine the presence and relative abundances of natural long-lived radioisotopes such as potassium-40, thorium, and uranium and induced radioisotopes on the surface of the moon.

Scientific significance: Information obtained can be compared with known relative abundance of natural radioisotopes on the earth to obtain additional clues to the origin and history of the moon and the solar system. Induced radioisotope information will indicate some of the elements on the moon.

Approach: Use is made of a scintillator gamma ray sensor multi-channel pulse height analyzer to obtain a measure of the flux and energy distribution of gamma rays from the moon which can be analyzed by newly developed analytical computer programs to identify the elements emitting the gamma rays and their relative abundances.

Requirements:

a. To minimize the background gamma rays from the spacecraft, the gamma ray sensor needs to be mounted at the end of a boom of as long a length as is practicable.

b. Ideally the aperture of the gamma ray sensor should be allowed to look in the direction of the lunar surface at all times. Partial viewing of the lunar surface as the spacecraft orbits the moon is acceptable.

c. Since variations of the gamma rays will result from varying concentrations of the radioactive elements over the surface of the moon, it is desirable that the measurements be made with the sensor at a nearly fixed distance from the lunar surface to avoid the introduction of an ambiguity due to the variation of gamma ray intensity with distance from the moon's surface. Ideally, a near circular orbit is required.

Weight - 28 lbs.
Power - 4 watts
Volume - 850 cu. in.
Information rate - 500 bits/sec.

2. INFRARED EXPERIMENT

Objectives:

a. Map lateral variations of the moon's surface temperature and surface temperature gradients across terminator.

b. Provide information about existence and distribution of minerals on the moon's surface.

Scientific significance: Further knowledge about the thermal properties and composition of material on surface of moon. Information can be directly correlated with pictures of regions scanned.

Approach: Use can be made of an infrared grating spectrometer to scan a broad band of wavelengths periodically at a high rate. The aperture of the instrument will encompass a finite area of the lunar surface.

Requirements:

a. Optical axis should be within ± 10 degrees of the local vertical when measurements are made.

b. Instrument should have unobstructed view of the moon.

c. Instrument must be shielded from the sun and isolated thermally from spacecraft as best as possible.

d. Near circular orbit desired.

e. If visual observations are not feasible to obtain at same time, orbit inclination should be selected to insure scanning of region photographed and mapped previously with photographic Orbiters.

Weight - 4 lbs.
Power - 8 watts
Volume - 1,000 cu. in.
Information rate - 550 bits/sec.

3. BI-STATIC RADAR EXPERIMENT

Objectives: Determine average radar cross-section, surface roughness correlation functions, altitude measurements, reflectivity, and dielectric properties of the lunar surface.

Approach: The spacecraft carries radar receivers to detect radar signals directly transmitted from high power radar transmitters on the earth, and the radar signals from these same transmitters after reflection at the lunar surface.

Requirements: The spacecraft configuration and attitude should allow for the mounting of the specified receiving antennas to provide for optimum reception of the signals.

Weight - 5 lbs.
Power - 2 watts
Volume - 100 cu. in.
Information rate - 2 KC
20 cps, SCO
40 cps, SCO

4. MICROMETEOROID EXPERIMENT

Objectives:

a. Investigate the distribution and determine the flux, momentum, and energy of micrometeoroids.

b. Determine the presence of lunar ejecta particles.

Approach: Micrometeoroid sensors will be used to measure the flux, momentum, and energy of particle incident from several directions.

Requirements: Unobstructed viewing in direction toward moon, radially away from moon, and at right angles to the normal to the moon's surface.

Weight - 27 lbs. (3 arrays)
Power - 1.5 watts
Volume - 8" x 12"/array (3 arrays)
Information rate - 200 bits/min.

5. SOLAR PLASMA EXPERIMENT

Objective: Study spatial and temporal variation of the flux and energy distribution of the low energy protons and electrons of the plasma.

Approach: Charged particle electrostatic analyzer or multi-grid faraday cups may be employed as sensors.

Requirements: Spinning vehicle desirable to provide full 360° sweep. If stabilized non-spinner is used, spacecraft needs to accommodate several identical sensors, one oriented along the lunar radius vector looking toward lunar surface, another looking along opposite direction. One or more detectors at several angles to lunar radius vector are desirable.

Weight - 12 lbs.
Power - 8 watts
Volume - 300 cu. in.
Information rate - 500 bits/sec., Sample rate - 0.09 sec.

6. MAGNETIC FIELD EXPERIMENT

Objective: Investigate magnetic field in vicinity of the moon.

Approach: Utilize one or more magnetometers to measure the intensity and direction of the magnetic field.

Requirements:

a. Mounting of magnetometers at end of boom soon after injection.

b. Magnetically clean spacecraft to insure field of one gamma at distance of 20 feet from spacecraft.

Weight - 12 lbs.
Power - 7 watts
Volume -600 cu. in.
Information rate - 100 bits/sec.

7. PHOTOMETRY/COLORIMETRY EXPERIMENT

Objectives: Determine variation of the photometric function and color of lunar surface material.

Scientific significance: Provide additional information about the small scale texture of the upper most layer of surface material as well as the relative ages of overlying material.

Approach: Use of photometer and a color wheel photosensing device. Correlate information with photographs of areas investigated.

Requirements:

a. Sensors should look toward lunar surface.

b. Sensors should be shielded from direct and reflected sunlight.

Weight - 4 lbs.
Power - 4 watts
Volume - 200 cu. in.
Information rate - 500 bits/sec.

8. X-RAY FLUORESCENCE EXPERIMENT

Objectives: Detect the relative abundance of iron and nickel on the lunar surface.

Approach: Use is made of proportional counters to monitor the X-ray emission from the sun which excites the iron and nickel atoms and to detect the X-ray fluorescence from the lunar surface. The signals are processed by a multi-channel pulse height analyzer.

Requirements: Two proportional counters must be mounted so that one is looking at the sun when the other is looking toward the moon.

Weight - 18 lbs.
Power - 2 watts
Volume - 640 cu. in.
Information rate - 250 bits/sec.
54,000 bits/sec.

9. RADIOMETER EXPERIMENT

Objectives: Determine lunar surface thermal gradients.

Scientific significance: From a measure of temperature gradients some information about the layering of material on the moon's surface can be obtained.

Approach: Two or more microwave radiometers operating at discrete frequencies monitor the intensity of the signals. Those signals originate at different depths beneath the surface thereby giving an indication of the temperature as a function of depth.

Requirements: The antennas of the radiometers should be pointed in the direction of the lunar surface.

Weight - 6 lbs.
Power - 5 watts
Volume - 500 cu. in.
Information rate - 500 bits/sec.

10. SELENODESY EXPERIMENT

Objective: Determine the shape or figure of the moon, the distribution of mass within the moon, and the gravitational field of the moon.

Approach: Use is made of the very accurate range and range-rate tracking data to determine the short and long period perturbations in the orbital elements of a Lunar Orbiter. Use is made of sophisticated computer programs to obtain the desired information.

Requirements:

a. No special instruments or hardware are required.

b. Orbits should be elliptical (approximately .1 to .2 eccentricity) and have inclination angles covering a range from about 20° to 60° with emphasis on 30° to 60° inclinations.

c. Tracking for 2 to 3 orbits per day is required during the first month after injection of the spacecraft into orbit about the moon. After this initial period, the tracking cycle will be 2 to 3 orbits twice a week during the remainder of the active lifetime of the spacecraft.

Weight - none
Power - none
Volume - none

APPENDIX C

RESULTS OF LITERATURE SEARCH ON EXPERIMENTS

A. Title: Gamma Ray Spectrometer

B. Objective: To determine the elemental composition of the lunar surface by analysing the gamma ray spectrum emanating from the lunar surface.

C. Functional Description:

Several attempts have been made to predict the nature of the nuclear-radiation emanating from the lunar surface. Crude calculations have been made of the flux and spectrum of the gamma radiation from natural and induced radiation assuming various models. Although none of these predictions suggest high radiation levels, it is conceivable that this radiation may give an early clue as to the elemental composition of the lunar surface, the extent of differentiation which may take place in the evolution of the moon. If sufficient radiation exists a crude mapping of the elemental composition and a crude mapping of the ages of various lunar features may be possible.

It would seem that this experiment might consist of an exploratory phase (one flight) to see if a sufficient level is present for pulse height analysis and subsequent flights of more sophisticated apparatus if such is the case.

D. Functional Elements:

1. Sensor: Scintillation counter with phoswitch, anti-coincident or pulse shape discrimination.

2. Electronics: Power supplies, amplifiers and pulse height analyser.

E. **Mission Requirements:**

1. Circular, polar low altitude orbit stability

   $\sim 1^{\circ}$ 1 sec

2. Continuous for one month duration

3. No solar illumination constraints

4 & 5 See E.1

6. Correlation with high energy solar particle events

F. **Experimental Parameters**

1. Measure gamma ray spectrum

2. Gamma ray counter efficiency

   1 for $.1 < E_{\gamma} < 10$ Mev

3. See E.2

4. Frequency Response

5. Power Requirements ~ 4 watts

6. Environmental Requirements

   a. Sensor: Constant Temperature

      Somewhere between -20° and $+20^{\circ}$ C

   b. Electronics: 0° C to 50° C

7. Mounting Requirements:

   It would appear that gamma ray background induced by vehicle is negligible for galactic background. Solar particle events will cause serious problems. However, it is not apparent that a boom is required, except for possible desire for orienting detector toward moon.

8. Underestimated view of lunar surface

9. Physical Deminsions

a) Sensor: 6" dia. x 12" long 10 pounds

b) Electrons: 6" x 4" x 12" 8 pounds

10. Sensor and Electronics may be separated

11. Data rate: 500 wts/sec

Title: Infrared Experiment

Objective:

a) Map lateral variations of the moon's surface temperature, and surface temperature gradients across terminator.

b) Provide information about existence and distribution of minerals on the moon's surface

Functional Description:

a) Mechanically moved scanning mirror projects infrared radiation from successive points on the lunar surface onto a radiation cooled radiometer which measures the energy with wavelengths primarily in the vicinity of 4 microns. Mirror scan is in one direction, perpendicular to motion of satellite, covering a continuous band around the moon.

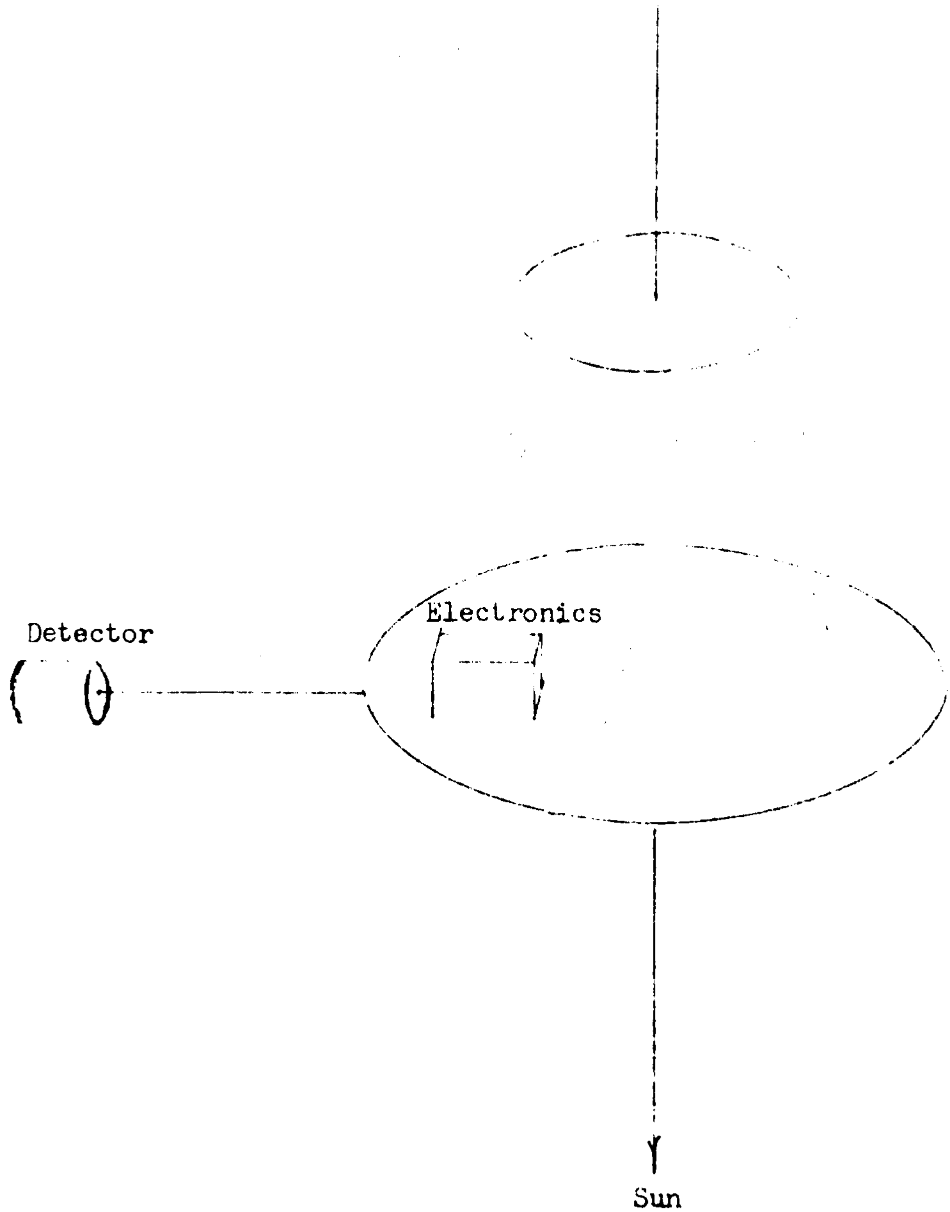
b) To obtain spectral data concerning materials, the scanning mirror is rotated into the direction of satellite travel and is synchronized with ground speed to provide a fixed image for a second or more. During this period the optical path passes through a Michelson interferometer with a moving mirror scanning through a wavelength band such as 2 to 15 microns.

Functional Elements:

a) Scanning radiometer is basically the High Resolution Infrared (HRIR) equipment of the Nimbus satellite, and manufactured for NASA by I.T. & T. Laboratories. It includes the sensor, with moving mirror assembly, lenses or mirrors, and PbS detector; and a magnetic tape recorder to store data for radio transmission.

b) Interferometer spectrometer is similar to Block Engineering Model 6, developed for Air Force Cambridge Research Center. An improved version, developed by the University of Michigan, may be available now. Integrating the two instruments into one will require some development.

[illegible]


Electronics
Detector
Sun

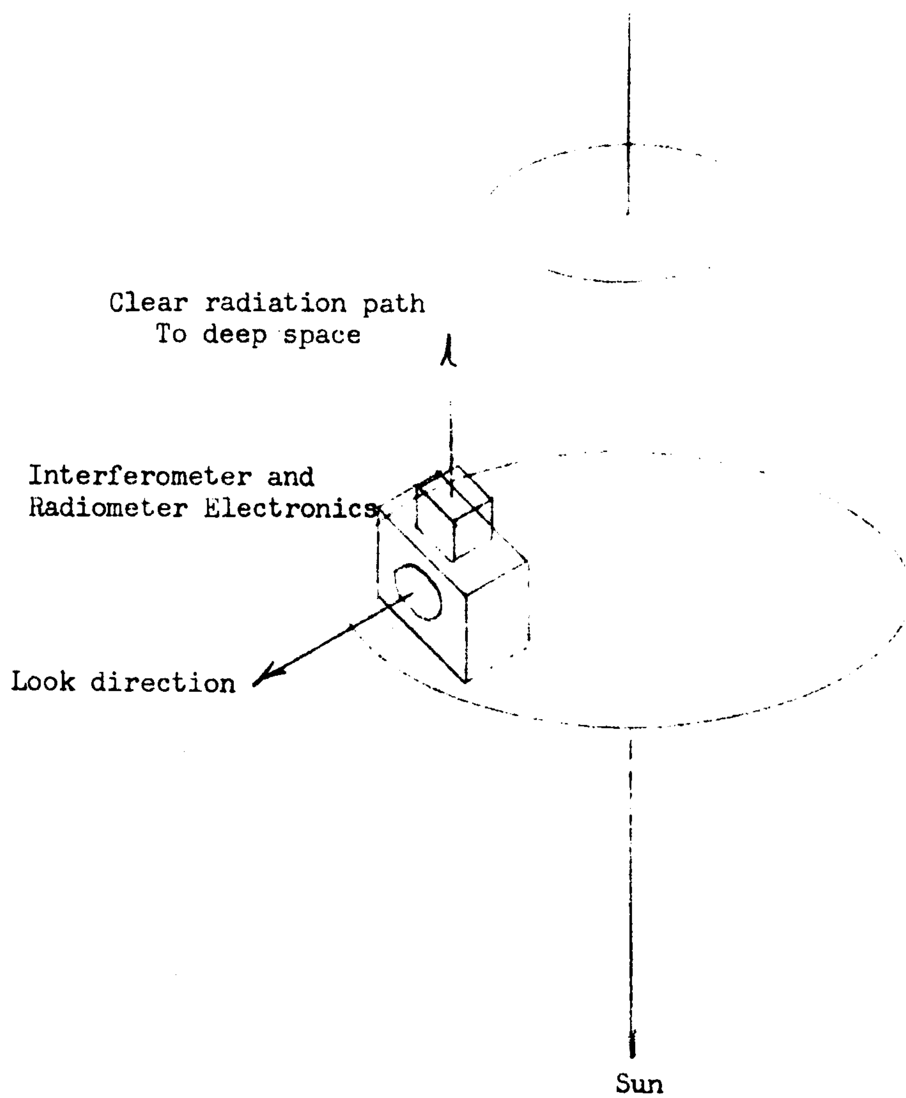
E. Mission Requirements:

1. Experiment Attitudes and Stability: sensor must point to lunar surface. Similar to photographic mission but stability tolerance is probably less stringent.
2. Experiment time duration:
   a) Experiment is useful at any time, day or nightside, but particularly of concern near the terminator.
   b) Lifetime requirement: as long as possible until complete lunar coverage is obtained
3. Solar illumination constraints: none except as noted under (2.) above.
4. Altitude constraints: as low as feasible.
5. Inclination constraints: none. polar orbit needed for full lunar coverage.
6. Correlation with other experiments. Correlations with medium resolution camera is required of all electromagnetic spectrum sensors unless the vehicle ephemeris and pointing can be known accurately enough to match points on the lunar surface with maps and photographs from other missions.

F. Experimental Parameters:

1. Measurements
   a) Radiance at 3-4 microns as f(t) during scan
   b) Spectral radiance during spectrum scan of fixed point.
2. Sensitivity a) 1° K
   b) 2×10^{-7} watts cm^{-2} $micron^{-1}$
3. Dynamic range
   a) $150-450^{\circ}$ K
   b) 2×10^{-7} to 10^{-4} watts cm^{-2} $micron^{-1}$
4. Frequency response
   a) 200 cps
   b) wavelength: 20 to 15 microns
5. Input power required 15 w
6. Environmental requirements:
   a) Sensor - free space (radiation cooling provided)
   b) electronics - free space

7. Mounting requirements. sensor on surface - looking face of vehicle.
8. Unobstructed field of view requirements: $\pm 45^{\circ}$ from nadir
9. Physical dimensions
    a) sensor: 12" x 12" x 12", 18 lbs
    b) electronics and recorder: 4" x6" x 8", 5 lbs
10. Sensor and electronic separability- satisfactory
11. Output requirements
    a) Data processing and conditioning - self-contained
    b) Data rate: 1400 bits sec^{-1}
    c) Data storage requirements: recorder included


Clear radiation path
To deep space
Interferometer and
Radiometer Electronics
Look direction
Sun

A. <u>Title:</u> Solar Plasma

B. <u>Objective:</u> To quantitatively measure the energy, flux, direction, and time variation of the solar plasma in the vicinity of the moon.

C. <u>Functional Description:</u>

Considerable evidence exists today that there is a flow of electrons, protons and alpha particles outward from the sun known as the solar wind. It is known that the flux of protons is between $10^7 - 10^9 p\ cm^{-2} sec^{-1}$ with energies in the 1 - 10 kev range. The flux and energy are functions of solar activity. Less is known about the electrons of the plasma and it has been speculated that alpha particles are present in the wind. Much of these data about the undisturbed solar wind has been obtained from plasma probes on Mariner II and Explorer XVIII (IMP).

From measurements on IMP and various Discoverers, the results of the interaction of this wind with the geomagnetic field have been partially determined. Evidence of a similar interaction of the wind with the moon has been suggested by measurements on IMP in which a perturbed magnetic field has been associated with a "lunar wake."

Plasma probes aboard the Lunar Orbiter will expand our information about the solar plasma, particularly in terms of particle nature, energy spectrum and direction. In addition, the nature of the interaction of the solar wind and the moon will be investigated. This information correlated with magnetic field measurements will establish the magnitude and nature of the lunar magnetic field.

D. <u>Functional Elements:</u>

1. Sensor: Plasma cups (three or four)

2. Electronics: a) Modulator grid power supply
oscillator, modulator high voltage supply
b) Timer, sequencer and encoder

E. <u>Mission Requirements;</u>

1. Experiment attitude and stability
less than 1° and 1°/second
2. Experiment time duration
continuous, one to three months
3. No solar illumination constraints--one solar plasma detector mounted so that it faces the sun direction
4. Elliptical orbit; say perigee at 40 n. mi. and apogee equal to perigee of anchored IMP. Primary interest in equatorial or low inclination orbit; secondarily in polar or high inclination orbit.
5. Correlate with magnetic field experiments and high energy particle experiments

F. <u>Experimental Parameters:</u>

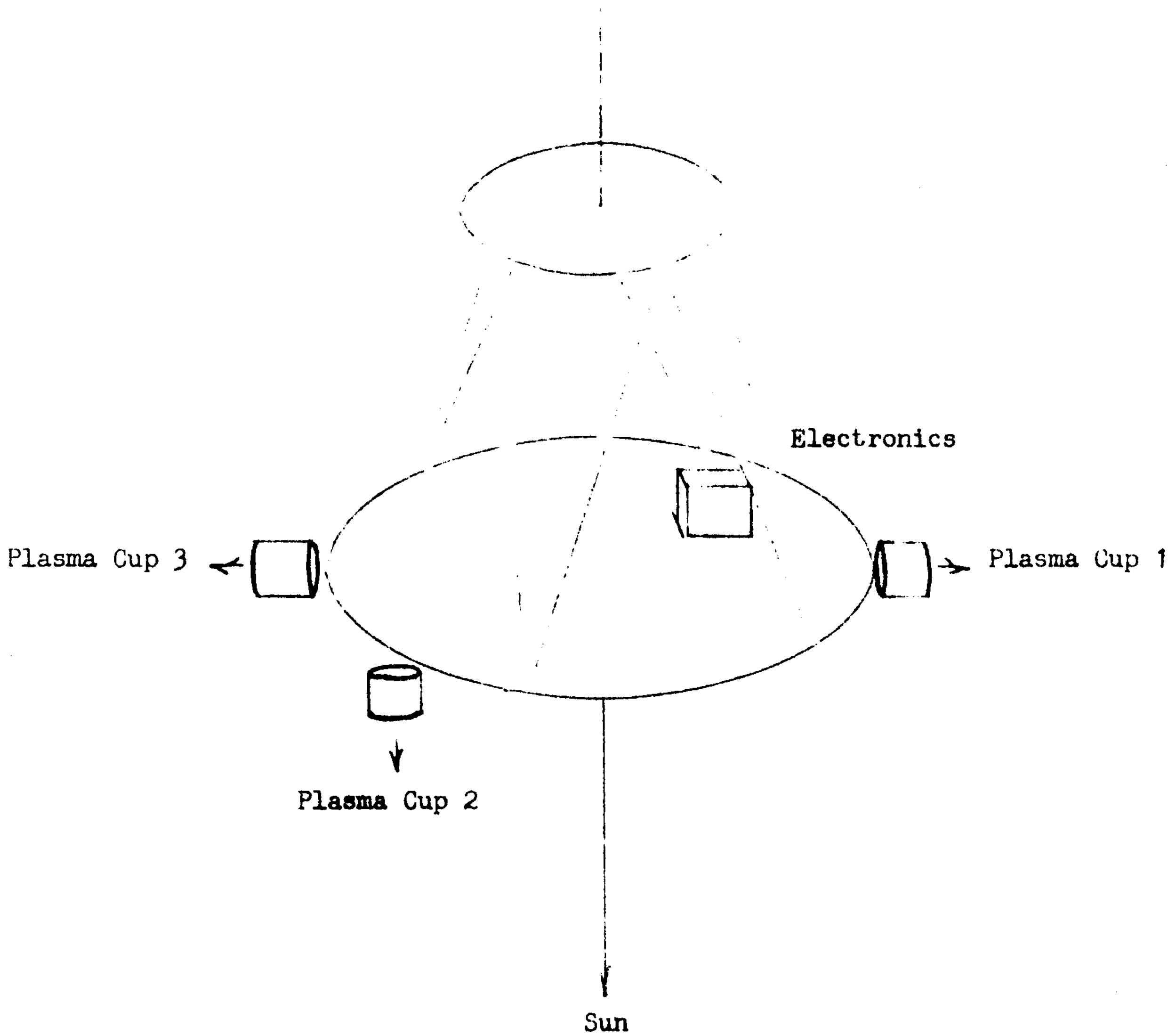
1. Proton, electron and alpha particle flux, energy spectrum, and direction.
2. 32 energy bands, over energy range of 30 - 10,000 electron volts
3. Particle flux range $10^4 - 10^{10}$ particles/(cm^2 sec.)
4. Frequency response ?
5. 4 watts power
6. Environmental response
   a) sensor temp. 100° C
   b) electronics 0° temp 50° C
7. Mounting Requirements
   a) one sensor pointed toward sun direction
   two sensors 90° from sun direction in opposite direction (180°). Must be outside thermal shroud.
   b) electronics on vehicle base plate
8. Unobstructed field of view
9. Physical Dimensions

a) sensor	2 pounds each	6" diameter	4" deep
b) electronics	5 pounds	6" x 6" x 6"	

10. Sensor and electronics may be separated by several feet but will require high-frequency coaxial cabling

11. Output Requirements

Storage would be required for time behind moon. During scan, 32 channels per minute (each sensor sampled sequentially) with 16 increments per channel give a bit rate of 128 bits per minute.


Electronics
Plasma Cup 3
Plasma Cup 1
Plasma Cup 2
Sun

Title: Magnetic Field Measurements

Objective: Measure the magnitude and direction of the lunar magnetic field.

Functional Description:

Recent measurements made with magnetometers on Explorer XVIII (IMP) have mapped the interplanetary magnetic field and have defined the magnetohydrodynamic shock wave of the Earth and its magnetosphere. In addition, some observations have been interpreted to indicate a magnetohydrodynamic wake of the moon. The implication of this wake is a possible magnetic field of the moon. The question of whether this is an intrinsic or an induced magnetic field has not been answered theoretically or experimentally. So far, the direct measurements made by Lunic II indicate that the field, if it exists, is less than 30 gammas at 55 km from the lunar surface.

Since the interplanetary field is assumed to be in the order of 2-4 gammas in magnitude in the vicinity of the moon with a direction approximately 5^{o} - 10^{o} from the sun line, a vector magnetometer having a sensitivity in the range of 1 - 100 gamma would be appropriate for the Lunar Orbiter (min. altitude = 40 n. mi.)

A significant observation to be made, in addition to the magnitude and direction of the lunar field, is a survey for possible turbulent fields of the "lunar wake" and the interaction of the earth wake with the lunar magnetic field. The latter two measurements require appropriate orbital considerations. The significant requirement for all the magnetic measurements is that the magnetometer be isolated from vehicle-induced fields. This requirement puts an upper limit on the vehicle-induced field of about 1-2 gamma at the location of the magnetometer and, in turn, severely restructs vehicle design, manufacture, and payload.

D. Functional Elements:

1. Sensor: Helium vapor magnetometer (Mariner IV) measuring B_x, B_y and B_z.
2. Electronics: current supply for coils, sequencer, amplifier, and 50 Mc RF oscillator.

E. Mission Requirements:

1. Know the orientation of vehicle within $\pm 1^\circ$ and be stabilized to 1°/sec
2. Continuous, one to three months
3. No solar illumination constraints
4. Prefer elliptic orbit: 40 nautical miles perigee to apogee equal to perigee of anchored IMP. First equatorial or lower inclination orbit, then solar high inclination orbit.
5. Correlate with plasma experiment.
6. Magnetic field background (vehicle) less than a few gamma (1 gamma (γ) = 10^{-5} gauss); see experiment functional description for interplanetary field.

F. Experimental Parameters:

1. Measurements of magnetic field in three perpendicular axes.
2. Sensitivity: $\pm .5\gamma$
3. Dynamic range : 0 - 200
4. Frequency response ?
5. Power input 7 watts
6. Environmental requirements
   sensor & electronics -20 to 55° C
7. Mounting Requirements
   The mounting requirements (length of boom) will depend on the magnetic field of the spacecraft. Since the spacecraft was not designed for conducting magnetic field measurements, it is very likely that such measurements cannot be carried out on the spacecraft; i.e., the boom may exceed space system capabilities.

8. No field of view requirements

9. Physical Dimension

   sensor and electronics 150 cubic inches

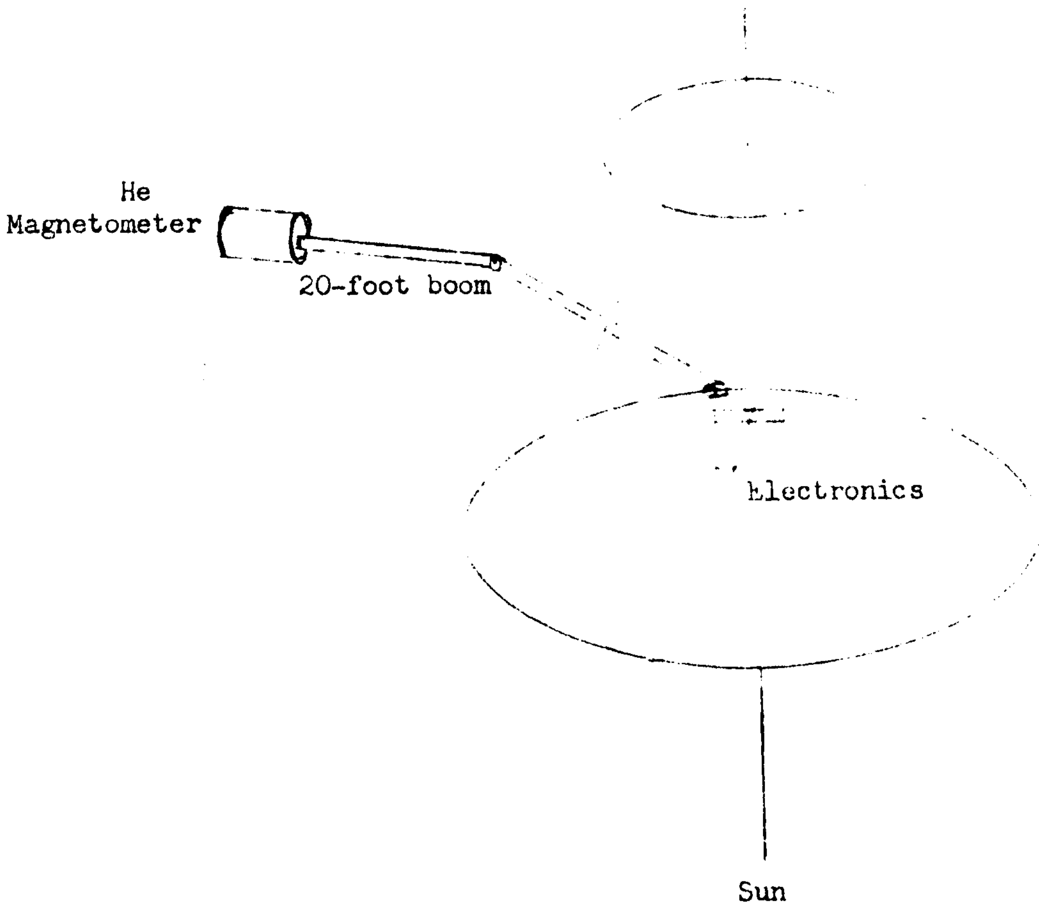
   weight 7.5 pounds

10. Sensor and electronic separability not desirable

11. Output requirements

    data rate >30 bits/second

    with storage for time behind moon


He
Magnetometer
20-foot boom
Electronics
Sun

A. **Title:** X-Ray Fluorescence

B. **Objective:** To measure the Ni and Fe content of the lunar surface.

C. **Functional Description:**

There is a possibility of detecting the presence of exposed iron and nickel minerals on the lunar surface through measurements of solar-X-ray-induced X-ray fluorescence. Considerable information on the fluorescence of various minerals has been obtained in the laboratory. The K, L, and M shell series have been measured for iron and nickel. The known X-ray emission by the sun during disturbed periods suggests that detection of X-ray fluorescence of the lunar surface is feasible. A study will be required to determine the frequency ranges excited by the solar X-rays.

Two detectors would be required to carry out this experiment. One, monitoring the sun, measures the intensity and frequencies impinging on the lunar surface; and a second measures the intensity, frequency, and location of induced radiation emanating from the lunar surface.

D. **Functional Elements:**

1. Sensors:

   One sensor geiger counter operating in proportional region to monitor solar X-ray spectrum. One or more sensors to analyze X-ray spectrum from lunar surface. The sensor(s) looking at lunar surface should consist of a set of slits, a crystal, and a geiger counter operating inthe proportional region.

   Electronics:

   High voltage regulated power supply amplifiers and pulse height amplifiers.

E. **Mission Requirements:**

1. Circular orbits at low altitudes and at low inclinations. Stability .01 degree/sec. (?)
2. Continuous so that background (cosmic radiation and other noise sources) could be determined.

3. Solar illumination required during measurements.

4. See E. 1.

5. See E. 1.

6. Could be correlated with high energy particle experiment.

F. Experimental Parameters:

1. Measure incident and reflected X-ray spectra.
2. Sensitivity ? depends on outcome of study.
3. Dynamic range: K, L, & M series for Ni and Fe $\sim$0.8 to 10 Kev
4. Frequency response: $\sim$1 to 20 Å (wavelength response)
5. Power requirements: $\sim$5 watts
6. Environmental requirements

   sensors:

   a) sun monitor: $0^{\circ}C < T < 100^{\circ}C$

   b) X-ray spectrum analyser: $20^{\circ}C \pm 5^{\circ}C$

   c) electronics: $0^{\circ}C < T < 50^{\circ}$
7. Mounting

   a) Solar monitor looking in sun direction

   b) Spectrum analyser looking at moon
8. Undisturbed view of moon and sun
9. Physical Dimensions

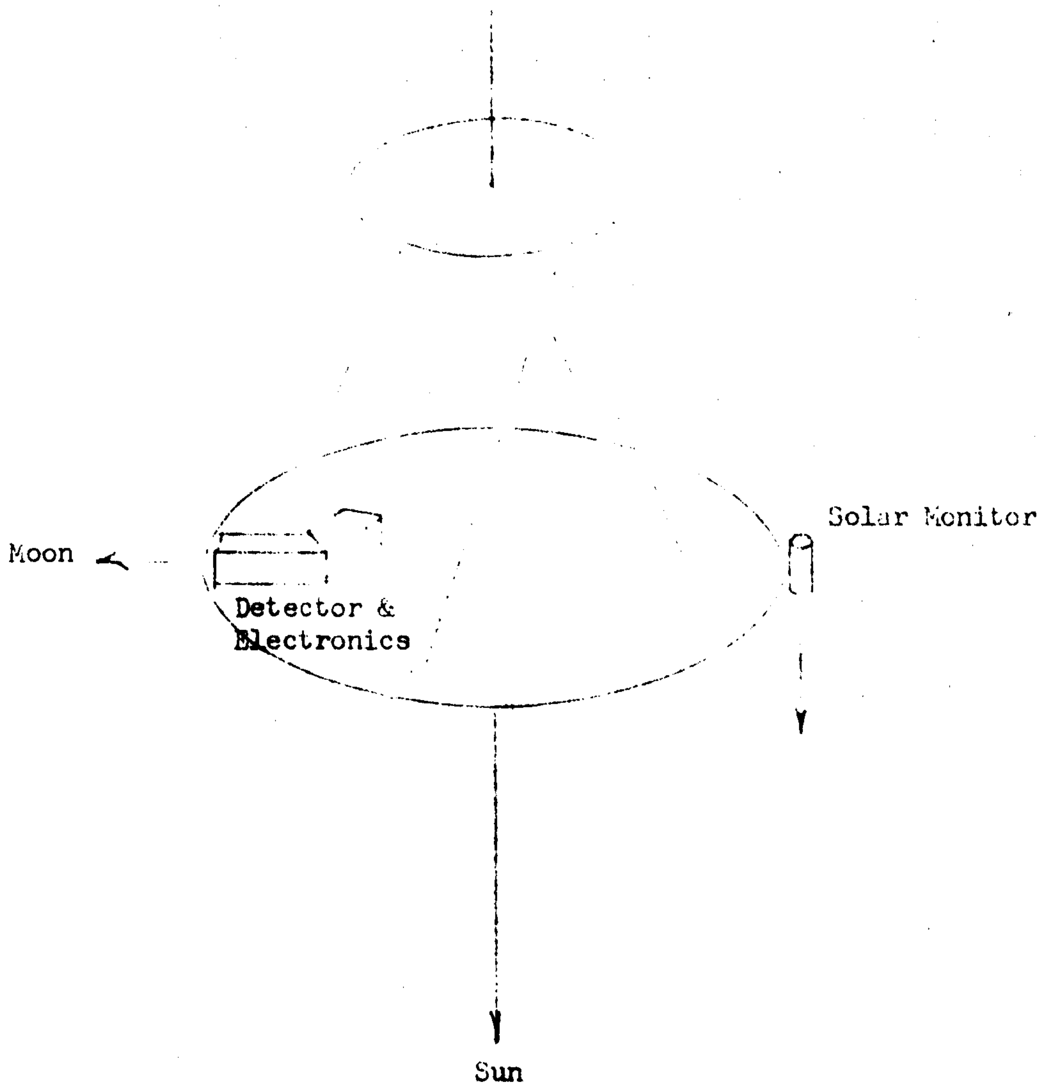
   a) Sun monitor: 2" dia., 6" long, 1 pound

   b) X-ray spectrum analyser: 300 cubic inches, 10 pounds

   c) Electronics: 250 cubic inches, 5 pounds
10. Sensor and electronics separability: Solar monitor can be separated from X-ray spectrum analyser and each from the electronics but high voltage cable must be provided to each.
11. Output requirements:

    data rate: 250 bits/sec to 54,000 bits/sec.

    Data rate behind moon will be less and storage required only for this amount.



Solar Monitor
Moon
Detector &
Electronics
Sun

A. Title: Microwave Radiometry of Lunar Surface

B. Objective: Determine temperatures at various depths by use of different microwave frequencies

C. Functional Description:

Several experiments have detected various long-wave-length radiation from the moon. These measurements provide some idea of the thermal gradient below the surface.

Extension of these measurements to frequencies which cannot be used for earth-based experiments because of atmospheric absorption and scattering should be conducted on the Lunar Orbiter. Such measurements along with the greater accuracy allowed by the proximity to the lunar surface should provide the basis for a better understanding of the thermal gradient and its time variation.

The power radiated from the moon depends on its temperature and emissivity. The temperature, measured when an antenna looks into a semitransparent and lossy medium, will be determined by the temperature at approximately one skin depth below the surface. Thus, a measurement of temperature at different microwave frequencies can give the temperature at varying depths near the lunar surface since the skin depth is inversely proportional to the square root of the frequency.

A crystal video-type receiver with the standard Dicke comparison radiometer circuit was flown on Mariner 2. This radiometer operated at two wavelengths of 13.5 and 19 mm, chosen because of expected transparency or opacity of the Venusian atmosphere. A parabolic antenna of 48.5 cm (19") diameter was required to obtain the desired resolution of Venus from the expected miss distance of the order of 10,000 to 30,000 mi. A somewhat similar system is appropriate for the Lunar Orbiter. A wider survey of frequencies is desirable, but less angular resolution is necessary because the LO will be much closer to the surface. Thus, center frequencies of 3 mm , 1 cm, and 3 cm would be chosen with either 3 mm and 1 cm or 1 cm and 3 cm on any one vehicle. The emission at these frequencies will be picked up with a 10-inch parabolic antenna and converted to temperature with a Dicke radiometer. Reference horns looking at free space will provide a calibration signal. The system design follows the Mariner 2 experiment with appropriate modifications for change of frequency and angular resolution.

D. Functional Elements:

1. Scanning parabolic antenna
2. Reference horn antenna
3. Receiver (crystal or superhet with klystron local oscillator)
4. Dicke radiometer
5. Power supply and servo control

E. Mission Requirements:

1. Scanning parabola must see moon over illumination angles from vertical through terminator into dark side.
2. Parabola located on moon side of LO over reasonable fraction orbit
3. Stability 1° over periods of ~ 40 sec.
4. Experiment senses over $\sim 125^\circ$ arc per orbit
$$= \frac{125}{360} \sim 35\% \text{ of time}$$
5. Lifetime $\sim$1 month
6. Altitude $<$ 100 mi. desirable
7. Inclination - Polar or high latitude eventually provides more information
8. Correlate with other mapping experiments optical, X-ray, IR, etc.

F. Experimental Parameters:

1. Measure temperature

$$T_a = \frac{\text{Power received by antenna}}{k \quad \Delta f}$$

where k = Boltzmann constant

Δf = receiver bandwidth

2. Sensitivity at various positions and look angles

$$\delta T = C \frac{T_a + T_r}{\sqrt{B\tau}}$$

where C = constant of order of unity

B = predetection bandwidth $\sim 10^9$ c.p.s.

τ = receiver post detection averaging time ~ 40 sec.

T_r = inherent noise of receiver

Typically, $T_r \gg T_a$. Hence

$\delta T \sim 10^{\circ}$ is to be expected

3. Dynamic range

T_a from 100° K to 400° K

4. Frequency response?

5. Pwr. requirements. Avg. 4 watts. Peak 9 watts.

6. Environmental requirements

Temp $\angle 65^{\circ}$ C for electronics

Avoid meteoroid penetration of antenna

7. Mounting

Antenna located so that with 125° scan it can see the moon as much as possible

8. Field of view $\sim 6^{\circ} \times 6^{\circ}$ for 3 cm channel. $0.6^{\circ} \times 0.6^{\circ}$ for 3 mm channel

9. Physical dimensions

Parabolic Antenna	25 cm diameter, (10")	$\sim$5 cm deep (2")
Horns	$\sim$3 x 10 cm aperture (1.3" x 4")	10 cm long (4")

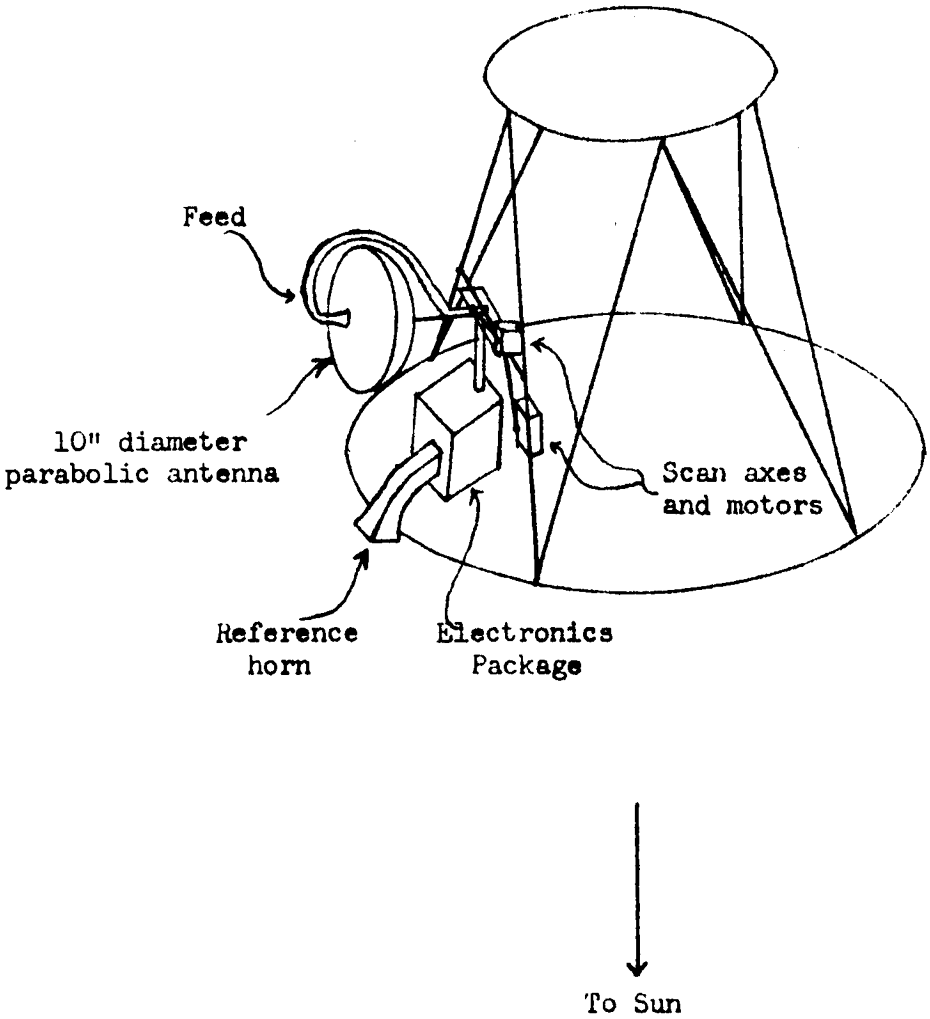
Electronics volume $\sim 1/3$ ft^3

Total weight ~ 20 lb

10. Detector should be mounted near antenna for min. loss

11. Output data rate $\sim \frac{5 \text{ bits}}{\text{measurement}} \times \frac{6 \text{ measurements}}{\text{min.}} \sim 0.5$ bits/sec

(+ position + angle data)


Feed
10" diameter
parabolic antenna
Scan axes
and motors
Reference
horn
Electronics
Package
To Sun

A. Experiment:

MICROMETEOROID EXPERIMENT

B. Experiment Objectives:

1. To investigate basic characteristics of cis-lunar and near-lunar meteoroids such as mass, spacial and temperal variances in particle flux and the mass, momentum and velocity of the incident particles.
2. To determine the presence and the above-named properties of lunar ejecta particles.

C. Experiment Functional Description

Particle sensors will be used to measure various properties of impacting particles. Data from these sensors will include particle velocities and the number of particles encountered having momenta larger than certain threshold levels. This information will be combined with data describing the location and orientation of the spacecraft to construct models of the meteoroid environment.

D. Experiment Functional Elements:

1. Sensor

The sensor consists of three detector arrays, one directed towards the lunar surface, one viewing radially away from the moon, and the third pointed at right angles to the surface normal. Each array consists of two independent detecting areas as shown in Figure 1. One, a standard acoustic impact detector, has a 2π steradian view of space and is used to determine the omni-directional flux of particles from the hemisphere it is sensing. The detector consists of a metallic impact or sounding board with a piezoelectric transducer mounted on the farside. The transducer is acoustically tuned to the mechanical vibrations induced in the plate during impact.

The second detector of each array consists of a thin foil laminate spaced several inches in front of another sounding board, and is used to measure particle velocities. Average particle velocities are determined by measuring the time between particle penetration of the foil and impact with the shielded sounding board. Penetrations of the foil are determined either by detecting the light flash produced during penetration or by charging the foil electrically as a capacitor and then sensing its momentary discharge at the time of penetration. For the dimensions shown in Figure 1, the maximum error in velocity measurement due to the particle flight path not being parallel to the sensor axis will be 14.3 percent. However, assuming no perferred particle source direction and that the particle passes through the sensor entrance window, over 70 percent of the impacting particles will have a flight path equal to or within the median of the maximum viewing angle. The velocity error associated with these particles will be 3.6 percent or less; for 90 percent of the impacting particles the flight path error will be less than 10 percent.

2. Electronics

The electronics for each impact detector channel will consist of one signal-conditioning amplifier and one or two monostable multivibrators and storage counters. Different detector sensitivity thresholds are obtained from various levels of amplification within the amplifier. The unshielded detectors will require two counters, one with a minimum capacity of 512 counts for high sensitivity, the other with a capacity of 256 counts. The shield impact plates of the velocity detectors will need only one counter with approximately

128 count capacity due to its reduced sensitivity and viewing angle. However, a counter will also be required to record the number of penetrations of the thin foil. Since this element will be penetrated by many smaller particles, the count storage for this sensing element will have to be larger, on the order of 4098 counts. The circuitry for this sensor will also include an amplifier and monostable multivibrator for each channel.

Velocity measurements are accomplished by gating the output of a 5-10 mc oscillator to a counter; the count is initiated by a signal from the foil penetration and stopped by a signal from the impact plate or a limiting timer in case the particle does not reach the plate. The method of data storage will depend upon the type of velocity data desired. The average velocity of all particles can be obtained from a counter that simply sums the output of the 10 mc counter, eliminating, of course, the spurious data from particles that puncture the foil but do not impact the plate. A more logical and perhaps the most practical method would be to divide the expected velocity range into several regions and record the number of particles for each velocity range. This technique will require one counter for each expected range. The ultimate would be to record the velocity of each individual particle, a technique that would require the use of a large storage core or a tape recorder. However, if data on the spactial distribution of meteoroids are to be gathered, this last method will be necessary.

E. Mission Requirements:

1. Experiment Attitude and Stability:

This experiment is not critically sensitive to spacecraft attitude; one axis of the craft must be pointed at the lunar surface.

2. Experiment Time Duration:

This experiment will operate continuously throughout the life of the spacecraft.

3. Solar Illumination Constraints:

None

4. Altitude Constraints:

None

5. Inclination Constraints:

None

6. Correlation Requirements with other experiments:

The location and orientation of the spacecraft will have to be known and recorded at the time of each impact if information on the spacial distribution of the particles is to be determined.

F. Experiment Parameters

1. Measurements:

   a. The number of particles encountered having momenta above a certain threshold value.

   b. The average velocity of a portion of the particles encountered.

2. Sensitivity:

   a. Momenta - 1.0, 0.5 and 0.1 dyne-sec.

   b. Velocity

      Maximum error @ 30 km/sec = 15%

      .72 probability error @ 30 km/sec = 4%

      .90 probability error @ 30 km/sec = 10%

3. Dynamic Range:

Not applicable

4. Frequency Response:

Velocity Measurement will require a 10 mc counter.

5. Power Input:

Impact Detectors - 6 channels @ 250 mw/channel - 1.5 watts

Velocity Measurement -

Photosensors - 3 channels @ 1.25 w/channel - 3.75

Capacitor - 3 channels @ 500 mw/channel - 1.5 watts

Tape Recorder 0.7 watts

Total System 3.0 to 6.0 watts

6. Environmental Requirements:

a. Sensor: - 50° F to 250° F

b. Electronics: 15° F to 120° F

7. Mounting Requirements:

Acoustic isolation from spacecraft

8. Unobstructed Field of View Requirements:

Nominal 2 π steradian unobstructed view in each of three directions.

9. Physical Dimensions:

a. Sensor:

3 arrays - 8" x 8" x 12" volume each.

5 pounds each

b. Electronics

1 package - 6" diameter x 8" high Volume

5 lbs. including protective case.

Tape Recorder - 3 pounds

10. Sensor and Electronics Separability:

No restriction

11. Output Requirements:

a. Data Processing and/or conditioning.

Input channel - each

1 signal conditioning amplifier and 2 monostable oscillators

Velocity channel - each

2 signal conditioning amplifiers and 2 monostable oscillators

Total - 9 amplifiers, 12 monostable oscillators

b. Data Rate:

Total - 200 bits/minute

c. Data storage Requirements

Impact Channel - each

1-512 bit counter

1-256 bit counter

Velocity channel - Average velocity - each

1-128 bit counter

1-4096 bit counter

1-32,768 bit accumulating counter

Velocity Channel - six velocity ranges - each

7-128 bit counters

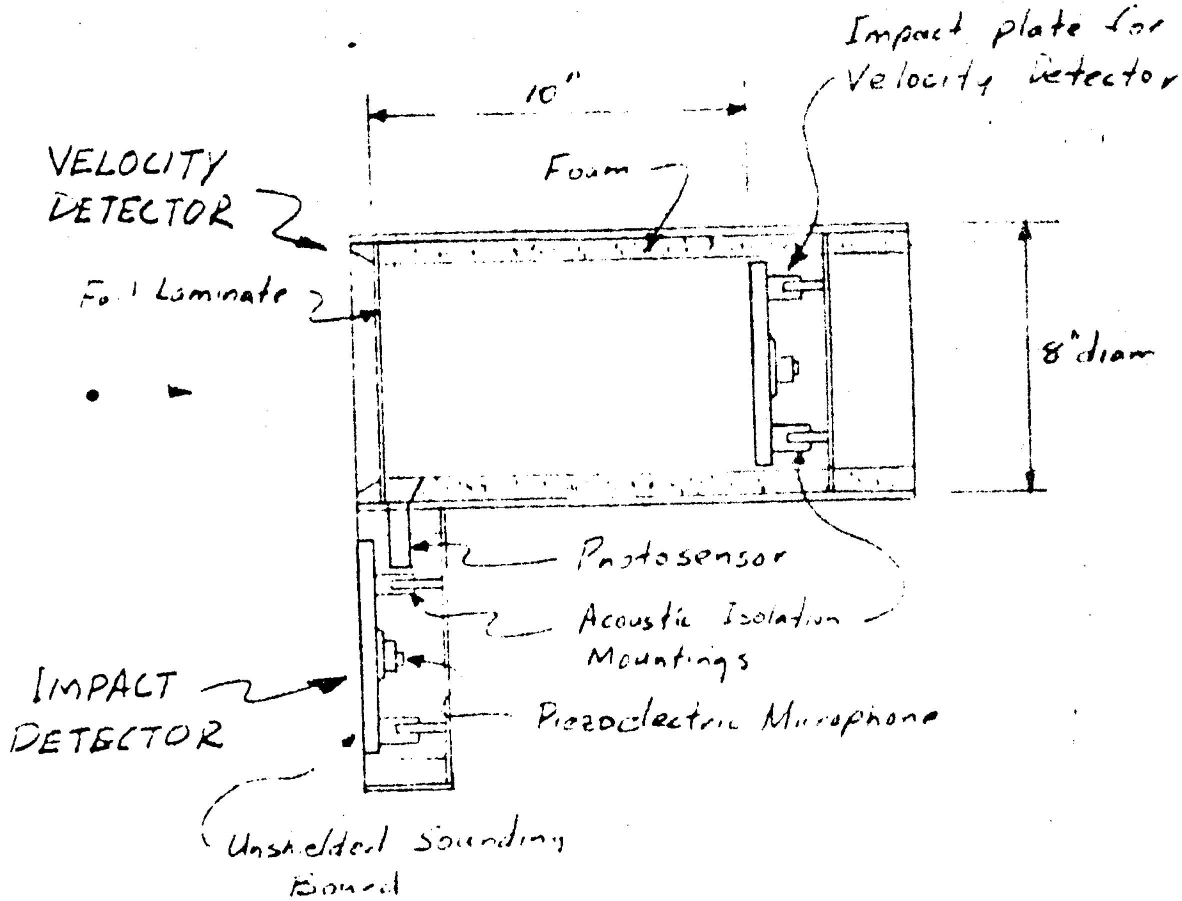
1-4096 bit counter.

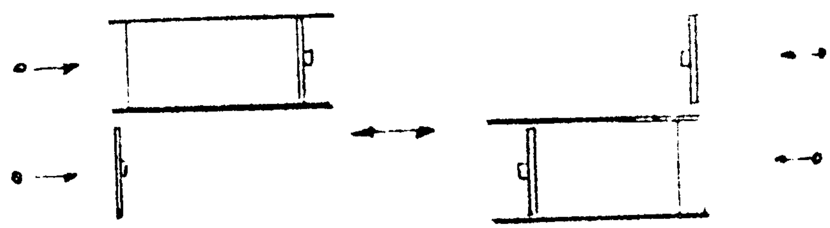
Velocity Channel - Individual Particle Velocities

1-128 bit counter per channel

1-4096 bit counter per channel

plus 1 tape recorder for the experiment





EXPLODED VIEW OF TWO DETECTOR ARRAYS

FIGURE 1 - METEOROID DETECTOR

A. EXPERIMENT: PHOTOMETER-COLORIMETER (PCE)

B. EXPERIMENTAL OBJECTIVES

1. Photometry

Current information on the photometric characteristics of the lunar surface, based on Earth based observations, is limited to average values for the surface in general and to recent observations of a limited number of specific locations. In all cases, the data provides an average over an area limited by resolution of the sensor used, and is at least greater than a square kilometer. The measurements made by Fedorets[1] have been the basis for expressing the photometric dependence on albedo, and the geometry of illumination, observation, and surface. Eimer [2] and Herriman, et al, of JPL derived a photometric function based on Fedorets measurement and the observations of Minnaert [4], Sharonov and Sytinskaya [5] and others, as a basis for the preflight determination of photographic requirements for the Ranger photographs. Later (1964) Willingham[6] re-evaluated the work of Fedorets on the basis of the more complete work of Sytinskaya and Sharanov, showing that appreciable error and inconsistencies occurred in the data, particularly under conditions of high angle illumination. Willingham showed that determination of a normalized photometric function by extrapolation to zero phase was particularly subject to error because of the extremely marked change in reflectance as zero phase is approached and because measurement at precisely zero phase cannot be obtained from Earth since lunar eclipse occurs. Sytinskaya and Sharanov had shown that while the Moon everywhere exhibits a strong back-scatter, the phase relationship varies appreciably for different locations. This was also observed by Wildey and Pohn[7].

While the lunar surface exhibits a remarkably uniform back-scatter characteristic, significant variation does occur, and uncertainties remain in the data. Photometric measurement of surface detail, and the reflectivity at zero phase is necessary. These measurements can be obtained from an orbiting spacecraft, which can increase the observational resolution, obtain measurements at zero phase and eliminate the difficulties of observation and measurement through the Earth's atmosphere.

Until lunar landing is achieved and surface measurements can be extended to more than a very limited area, knowledge of the detailed lunar topography must rely upon the interpretation of photographs. Slopes and surface irregularities are evident only as they modify the observed brightness and produce brightness contrasts with the surrounding area. Determination of such slopes requires, implicitly, knowledge of the reflectance characteristics of the surface in question, not only its albedo, but also its dependence on illumination angle, line of sight, and surface normal.

Photographs from unmanned spacecraft cannot provide precise data on absolute brightness of the lunar surface because of uncertainties in the film exposure originating in shutter error. Pre-exposed edge data provide good control of the modifications of image density relationships inherent in the processing, scanning, transmission, and reconstruction of the photographs. Direct photometric measurements will, however, provide the necessary calibration if made in conjunction with the photographs so that direct comparison of the brightness of specific areas under identical illumination can be made.

2. Colorimetry

Color or color differences on the Moon are slight and rarely can be detected by direct visual observations. Color differences have, however, been demonstrated by instrumental measurements. Wildey and Pohn[7] have made measurements of 25 locations using the UBV system of Johnson and Morgan[8]. They show a B-V variation ranging from 0.820 magnitude for Aristarchus to 0.919 magnitude for Proclus. Gehrels, et al,[9] have also investigated color variation on the Moon, and appear in general agreement with Wildey and Pohn although some differences are present. Gehrels has shown that color is somewhat phase dependent and that brightness is also apparently related to solar activity. He also shows that color differences are widespread and that the color differences occur across rather sharp, well defined boundaries rather than a gradual blending. This latter observation appears to be of major significance with respect to variability of the physical character of the surface since it implies a difference in composition, origin, formation, or age.

Color measurements will provide a significant contribution to the determination of surface composition and character. Maximum surface resolution and accuracy of spectral data compatible with sensor capabilities and limitations will enhance the value of the colorimetry by limiting the averaging effect over the surface and providing controlled spectral information.

C. EXPERIMENT FUNCTIONAL DESCRIPTION

The PCE shall measure the intensity of the reflected light in two (2) spectral bands within the visible spectrum, and the total reflected light within the visible spectrum (400 to 700 millimicrons) while scanning a strip on the surface extending to both sides of the orbit. The functions of scanning and data encoding shall be contained within the PCE and shall be synchronous with the spacecraft clock. A unique identifying marker shall be generated as part of the data channel output to mark the start of each cross track scan.

D. EXPERIMENT FUNCTIONAL ELEMENTS

All functional elements shall be included within a single package.

E. MISSION REQUIREMENTS

1) The required attitude and attitude stability will be a function of specific mission requirements. For most possible missions the existing attitude system is sufficient.

2) Experiment Time per Pass and Total Life

Depending on the values of sun angle and viewing angle over which data is to be obtained, the time per pass will range from a few minutes to about 100 minutes.

The lifetime should be sufficient to allow measurements at any point on the lunar surface within the latitude limits established by the orbital inclination, i.e., approximately 28 days in orbit.

3) Solar illumination constraints - none.

4) Altitude constraints - none: resolution will be limited by the data rate of 500 bits/second.

5) Inclination constraints - none.

6) Correlation Requirements with other Experiments:

The data must be correlated with the photographic data. This requires accurate time-tagging and a method of reconstructing the data into an image on film or a display (non-real time) for accurate correlation with the photographs.

F. EXPERIMENT PARAMETERS

1) Measurements

Three (3) channels of 8 bit binary coded logarithmic data representing the intensity of the reflected light in each of the spectral regions.

2) Sensitivity

Ten (10) ft.-lamberts

3) Dynamic Range

Ten (10) to 5,000 ft.-lamberts

3a) Accuracy $\pm 3\%$

4) Frequency Response

Twenty-one (21) measurements/seconds

5) Power Requirements

Four (4) watts (operating only)

6) Environmental Requirements

Present LO specifications, rate of change of temperature less than 10°/hour.

7) Mounting Requirements

Must accurately maintain alignment to IRU.

8) Field of View

Highly dependent on specific mission requirements, assume a cone 20° in diameter centered on present camera axis.

9) Physical Dimensions

Four (4) lbs., 200 cubic inches

10) Sensor and Electronics Separability

Not recommended.

F. 11) Output Requirements

a) Must receive timing pulses from spacecraft clock

b) Data rate - 500 bits/second

c) No data storage required if telemetry subsystem can support the 500 bit per second rate

Subject: Bi Static Radar Implementation for Lunar Orbiter Block II

References:

1) Beckmann and Spizzichino, "The Scattering of Electromagnetic Waves from Rough Surfaces", The MacMillan Co., 1963

2) J. V. Evans, "The Scattering Properties of the Lunar Surface at Radio Wavelengths", Lincoln Lab Rep. 30-004, 1961

3) Skolnik, M. I., "Introduction to Radar Systems", McGraw Hill Book Company, 1962

4) Dickey and Craig, "Bistatic Correlation Radar for Velocity Sensing in Spacecraft", Journal of Spacecraft, Vol. I, No. 5, pp. 508-512

5) D2-100310-1, "Communications Subsystem Analysis"

A. Experiment: Bi Static Radar

B. Experimental Objectives: The relative complex dielectric constant for Earth, ϵ_e^*, has been found to be

$$\epsilon_e^* = \frac{\epsilon}{\epsilon_o} - 60\, i\, \lambda\, \sigma$$

where ϵ is the dielectric constant, ϵ_o is the dielectric constant of free space, σ the conductivity in mho/meter, and λ the wavelength. The reflection coefficient for horizontal polarization, R_h, and vertical polarization, R_v, similarly is given for a smooth earth by

$$R_h^{+} = \frac{Y^2 \cos \gamma - \sqrt{Y^2 - \sin^2 \gamma}}{Y^2 \cos \gamma + \sqrt{Y^2 - \sin^2 \gamma}}$$

$$R_v^{-} = \frac{\cos \gamma - \sqrt{Y^2 - \sin \gamma}}{\cos \gamma + \sqrt{Y^2 - \sin^2 \gamma}}$$

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1. Fedorets, V. A., "Photographic Photometry of the Lunar Surface", Vol. 2, Public Kharkev Observatory, 1952

2. Eimer, Manfred, "Photography of the Moon from Space Probes", Jet Propulsion Laboratory Technical Report No. 32-347, 15 January 1963

3. Herriman, A. G., H. W. Washburn and D. E. Willingham, "Ranger Pre-Flight Science Analysis and the Lunar Photometric Model", Jet Propulsion Laboratory Technical Report No. 32-384 (Rev.) 11 March 1963

4. Minnaert, M., "Photometry of the Moon", "The Solar System" Planets and Satellites, Vol. III, pp. 213-248, The University of Chicago Press, 1960

5. Sharanov, V. V., and N. N. Sytinskaya, "An Investigation of the Reflecting Power of the Lunar Surface", pp. 114-154, Public Astronautical Observatory, Leningrad, 1952

6. Willingham, D., "The Lunar Reflectivity Model for Ranger Block III Analysis", Jet Propulsion Laboratory Technical Report No. 32-664, 2 November 1964

7. Wildey, Robert L. and Howard A. Pohn, "Detailed Photoelectric Photometry of the Moon", The Astronomical Journal, Vol. 69, pp. 619-634, 1964

8. Johnson, H. L. and W. W. Morgan, "Fundamental Stellar Photometry for Standards of the Spectral Type on the Revised System of the Yerkes Spectral Atlas", The Astrophysical Journal, Vol. 117, No. 3, pp. 313-52, May 1953

9. Gehrels, T., T. Coffeen and D. Owings, "Wavelength Dependence of Polarization III. The Lunar Surface", The Astronomical Journal, Vol. 69, No. 10, pp. 826-852, 10 December 1964

where the normalized admittance Y is equal to $Y = \sqrt{\frac{\epsilon_e^*}{\mu_e}}$, μ_e the magnetic permeability of the Earth (usually assumed equal to unity) and ψ is the grazing angle (complement of incident angle) of the incident wave relative to the reflecting surface. It is reasonable to assume that similar relations exist for the lunar surface. Unfortunately, prior to Lunar Orbiter, the only method available for determining the lunar dielectric constants, radar cross sections, and reflection coefficients, consists of analyzing the backscatter signal of Earth based monostatic radars.

The Lunar Orbiter can, without extensive redesign, be modified so as to measure the reflection and scattering coefficients. This data will permit the determination of the lunar dielectric properties, surface roughness characteristics or correlation functions, and possibly the radar cross section area as a function of height. In addition, this data can be used for verifying the validity of many of the existing theories and models assumed pertaining to the properties of the lunar surface.

C. Experiment Functional Description

Reflection coefficient: The reflection coefficient can be measured with a small low gain antenna pointed towards Earth and a fixed high gain antenna orientated towards the lunar or reflecting surface. This reflection coefficient may be expressed by:

$$(R_S)_{rms} = P_{rms} R_o$$

where $(R_S)_{rms}$ is the rms magnitude of the reflection coefficient, and P_{rms} is the coefficient accounting for the surface irregularities, and R_o is the reflection coefficient for a smooth surface. Both the direct and the reflected signal power will be transmitted to DSIF where their ratio (reflection coefficient) can be obtained. The directivity of the high gain antenna shall be used to separate the directed from the reflected waves.

Dielectric Properties: The dielectric properties of the lunar surface can be obtained from the reflection coefficient. For both normal incidence and reflection, the reflection coefficient as described in Section B is equal

$$R = \frac{\sqrt{Y}-1}{\sqrt{Y}+1}$$

Since the dielectric constant generally is a complex quantity dependent on frequency, it can be obtained from the above relation by making measurements at two discrete frequencies. This would require either two front ends or receivers in the satellite.

The monitored data for this experiment would be transmitted to DSIF over the high power S-band link. If more sophistication is desired, the dielectric properties can be obtained by monitoring the depolarization effect of an incident wave of known polarization. This would require a precise knowledge of the incident wave, and the measurement of both the reflected horizontally and vertically polarized waves.

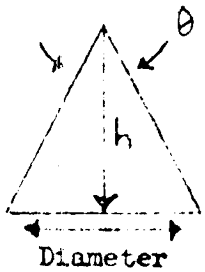
Surface Roughness: The surface roughness as a function of orbit can be obtained from the variations in the reflection coefficient as a function of orbit. However, the actual irregularities and surface electric properties can best be obtained by measuring the scattering or reflection coefficient for varying reflection angles from a fixed area on the surface. This would require rotating the high gain antenna in order to keep it oriented to a particular spot on the surface as the Orbiter moves in its orbit.

In order to ensure this function without undue requirements on the attitude control system, a two axis rotation capability for the high gain antenna might have to be added. This problem has already been investigated for use on the present Lunar Orbiter and found to be feasible.

In such a mode the scattering data would have to be both time and antenna angle tagged in order to enable accurate processing of the data at the DSIF. This data preferably would be continuous and would be transmitted in its monitored analog form to DSIF over the S-band high power-wide bandwidth mode.

Altitude Measurements: Altitude measurements can be obtained by use of a leading edge altimeter operating in a bi-static radar mode. The altitude would be proportional to the difference between the time of arrival of the leading edge of the direct pulse as received by the Earth directed low gain antenna, and the reflected pulse as received by the high gain antenna normally oriented to the lunar surface.

The diameter of the cone illuminated on the lunar surface for various high gain antenna beamwidths and orbit altitudes is given in Table I.



h, ORBIT ALTITUDE	ANTENNA BEAMWIDTH θ 2°	6°	10°
50 KM	1.75 KM	5.2 KM	8.75 KM
100 KM	3.5 KM	10.4 KM	17.5 KM
500 KM	17.5 KM	52 KM	87.5 KM

DIAMETER OF ILLUMINATED CONE

TABLE I

As shown in Table I, even for a narrow beamwidth, a large surface area will be intercepted. However, the size of the area covered can alos be limited by the duration of the transmitted pulse width. This, however, involves a trade study with the other bi-static measurements. A 20 pulse per second pulse repetition frequency appears satisfactory for the attitude measurements.

The altitude measurements will be telemetered back to Earth separately from the reflection coefficient measurements.

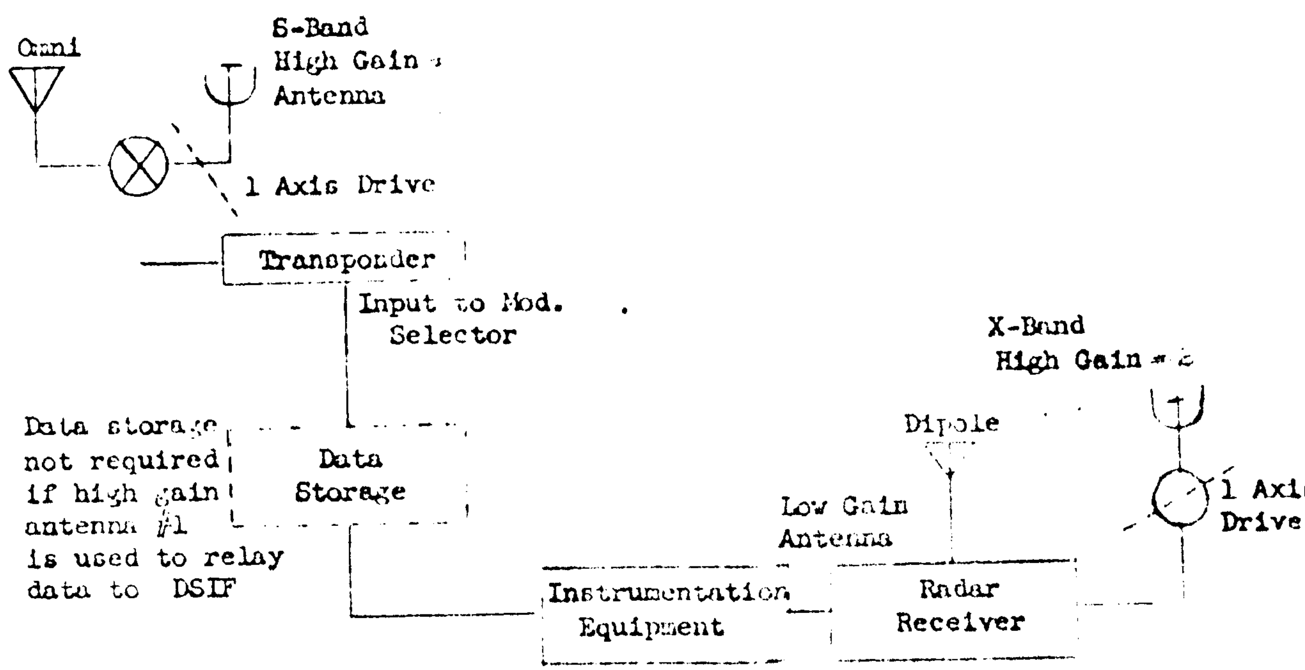
Radar Cross Section: The average radar cross section area generally is of concern when you are in the far field of reflector. Since the experiments will be performed at a perilune of 50-100 KM, which corresponds to the near field, the radar cross sectioned area can be obtained analytically by relating it to the reflection coefficient. In the far field this relation is

$$\sigma = R_o^2 \pi a^2$$

where a is the radius of the moon. In the near field, this relation has to be modified so as to take into account the surface area illuminated, antenna beamwidth, and surface roughness distribution function. This is thoroughly discussed in Reference (1).

D. Experiment Function Elements

It is preferable to separate the bi-static receiver and equipment from the command and telemetry systems. This is most easily accomplished by using different frequencies for each system. The radar experiment could use x-band (8 KMC) which would permit the use of facilities such as Lincoln Labs Haystack Radar, while the command and telemetry systems could simultaneously use the assigned 2 KMC Lunar Orbiter channel. Pulse radar probably would be used as it lends itself most easily to altitude measurements. Multitone cw radar could, however, be used. The maximum pulse rate is equal to the reciprocal of the time to illuminate the entire sphere or 85 pps if the entire surface is illuminated. Ideally, the altimeter measurements would use pulse transmission, while the reflection coefficient measurements could use continuous cw transmission.



Block Diagram for Transponder - Bi-Static Instrumentation Interface

Weight of Bi-Static Equipment:

Complete dipole assembly	1 lb.
High Gain antenna	
• Gear Drive (Optional)	3
• Boom	2-3
• Dish and Feed	2-3
	4-9 lbs.
Receiver and miscellaneous equipment	3 lbs.
Total	8-13 lbs.

If the high gain antenna is to be rotated about Z-axis in order to measure the scattering from a particular area on the surface, an additional antenna drive mechanism would have to be added to the boom. This would increase the weight by 2-3 lbs.

E. Mission Requirements

1. Attitude Requirements - The existing attitude system is sufficient for the bi-static radar experiments. If fixed antenna operation rather than 2-axis rotation is used, the attitude of the spacecraft, for each measurement - orbit pass, will have to be adjusted so that the boom is parallel to the lunar surface and is perpendicular to the orbit direction. This attitude restraint will permit all previously described measurement to be made, including scattering from a particular area as a function of reflection angle.

2. Experiment time per pass and total life - For a satisfactory operating altitude control system no minimum operating time per pass exists. The actual length of the mission will depend on the area selected for the measurements, but will be less than 28 days.

3. Solar Illumination Constraints - None, in fact the noise levels are less when the Moon is not solar illuminated, though this is not important.

4. Altitude Constraints - A perilune of 50 KM to 100 KM is required in order that the lunar surface appear as a plane to the incident and reflected waves.

5. Inclination Constraints - Generally, though not necessary, areas whose incident waves are normal to the surface will be selected. Measurements can not be made when the spacecraft orbit is such that the direct waves are within the beamwidth of the high gain antenna (with 5° of the horizon).

6. Correlation with Other Experiments - No direct connection exists with the other experiments though the data should be correlated with the results of the other experiments, i.e., photography and radiometry measurement.

F. Experiment Parameters

1. Measurements:

   a. Reflection coefficient measurements - two simultaneous measurements consisting of the direct and reflected signal strength.

F. 1. b. Altimeter Measurements - The time delay between the arrival of the direct and reflected pulse. This could be performed for each pulse though the delay time may be "smoothed" by averaging over several pulses in a servo system.

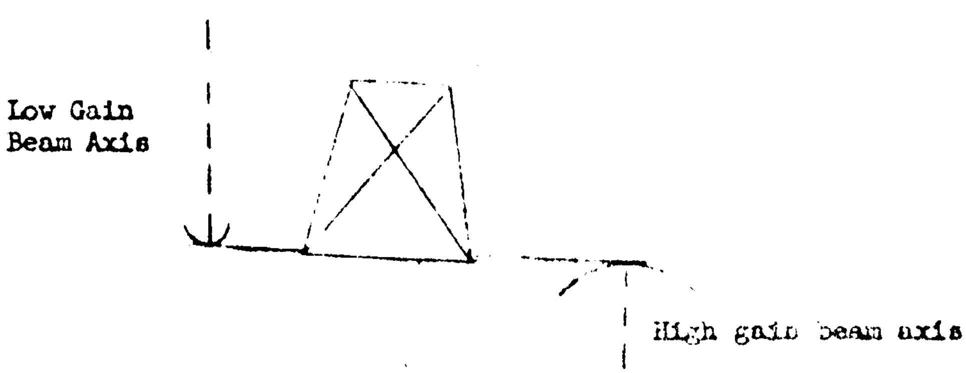
2. Sensitivity - As strong signals for both the direct and reflected waves can be expected a receiver noise figure of 10 db resulting in a noise density of -163 dbm per cps can be used. This results in a noise level of - 150 db in a 100 cycle bandwidth.

3. Dynamic Range - As the transmitted power can be varied (up to 100 KW) a linear radar receiver with a dynamic range of -70 to - 140 dbm would probably be satisfactory.

4. Power Requirements - As no spacecraft transmitter is required, 2 - 6 watts should be sufficient to operate the altimeter, radar receiver and measurement equipment. This does not include the power required to operate the S-band transponder and transmitter.

5. Environmental Requirements - Present Lunar Orbiter specifications.

6. Mounting Requirements - The electronics can be mounted similar to that in the present Lunar Orbiter. The high and low gain antennas should be mounted such that for normal incidence their beam axis are separated by 180°, as shown below:



Mounting or Separation of X-Band Antennas

Figure 1

F. 7. Directivity of Antennas - The X-band antennas shall have sufficient directivity such that the reflected and direct waves do not interfere with each other.

8. Physical Dimensions - Five to six pounds and 100-200 cubic inches for the receiver and electronics. In addition, from nine to twelve pounds for rotatable antennas are also required.

9. Sensor and Electronics Separability - See Item 6 above, for only requirements.

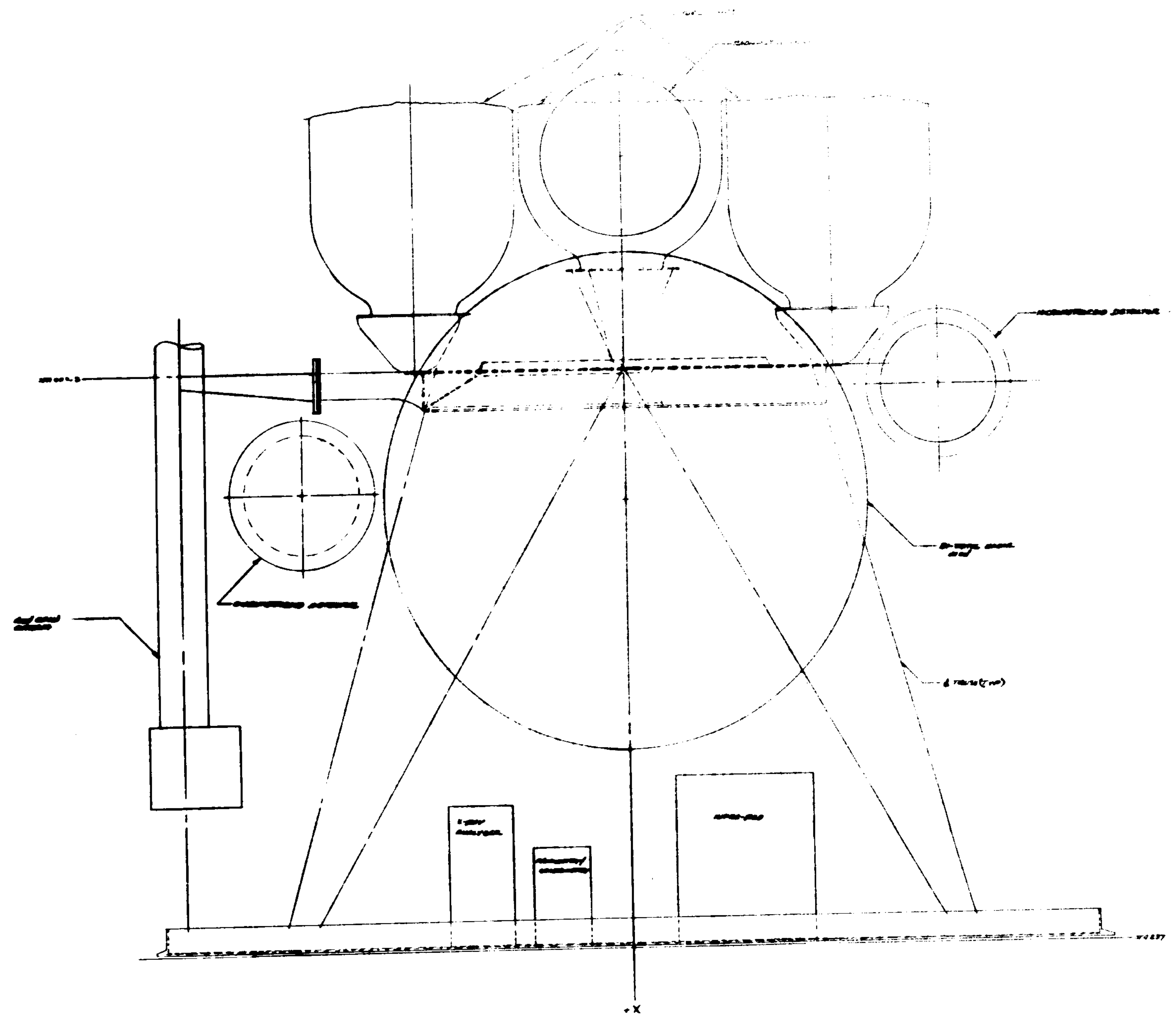
10. Output Requirements

    a. Separate analog signals for both direct and reflected waves and the necessary time and angle tagged data. This information can be transmitted to Earth over the present high power mode.

    b. Discrete averaged altitude data as determined by the bi-static radar altimeter. The present telemetry system is capable of transmitting this data to Earth.

11. Data Storage Requirements - None required if the present S-Band lunar orbiter high power transmitter can be used simultaneously with the proposed X-Band radar measurements. This will permit the relaying to Earth a 3 1/3 MC wide signal, providing ample room to place each measurement on its own subcarrier for processing on Earth.

12. Frequency - An X-Band radar frequency (approximately 7 - 10KMC) was assumed in order to provide isolation with the 2 KMC command and control system and to permit the use of radar facilities such as Haystack at Lincoln Lab. Several frequencies could be used for the experiments. Each frequency would require, however, a separate receiver or front end.



VIEW B-B

FIGURE 4.0.3.21

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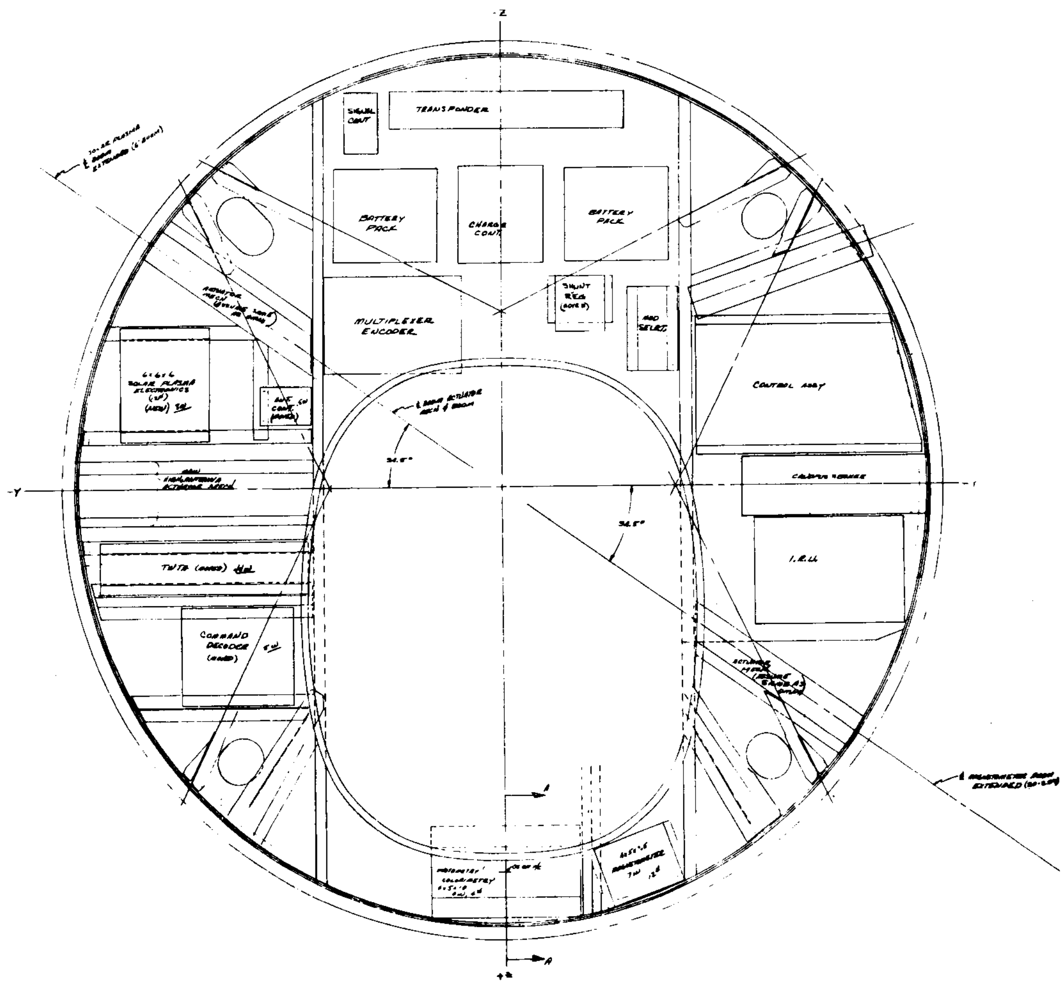
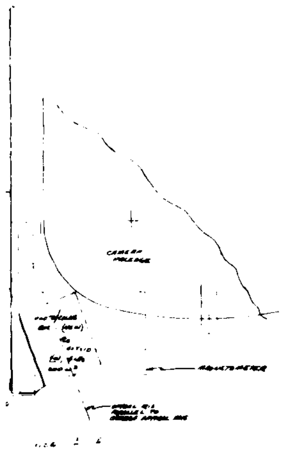
BOEING

LUNAR ORBITER ADAPTABILITY STUDY-CASE IV

LO-JWR-4

CASE Ia
ADDED EXPERIMENTS:
1. PHOTOMETRY/COLORIMETRY
2. MAGNETOMETRY
3. SOLAR PLASMA





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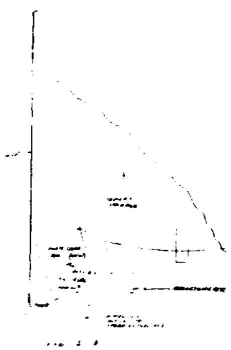
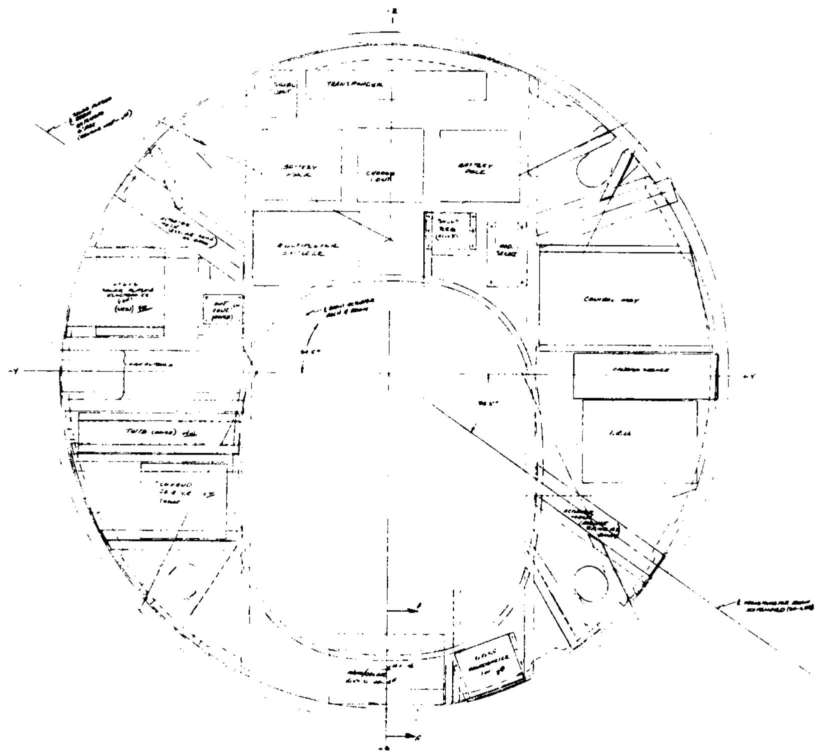
LUNAR ORBITER CAPABILITY STUDY CASE Ia
LO-JWR-1





ADDED
EXPERIMENTS

NOTE:

FIGURE 4.0.3.12

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LUNAR ORBITER
ADAPTABILITY STUDY
CASE Ia

LO-JWR-8

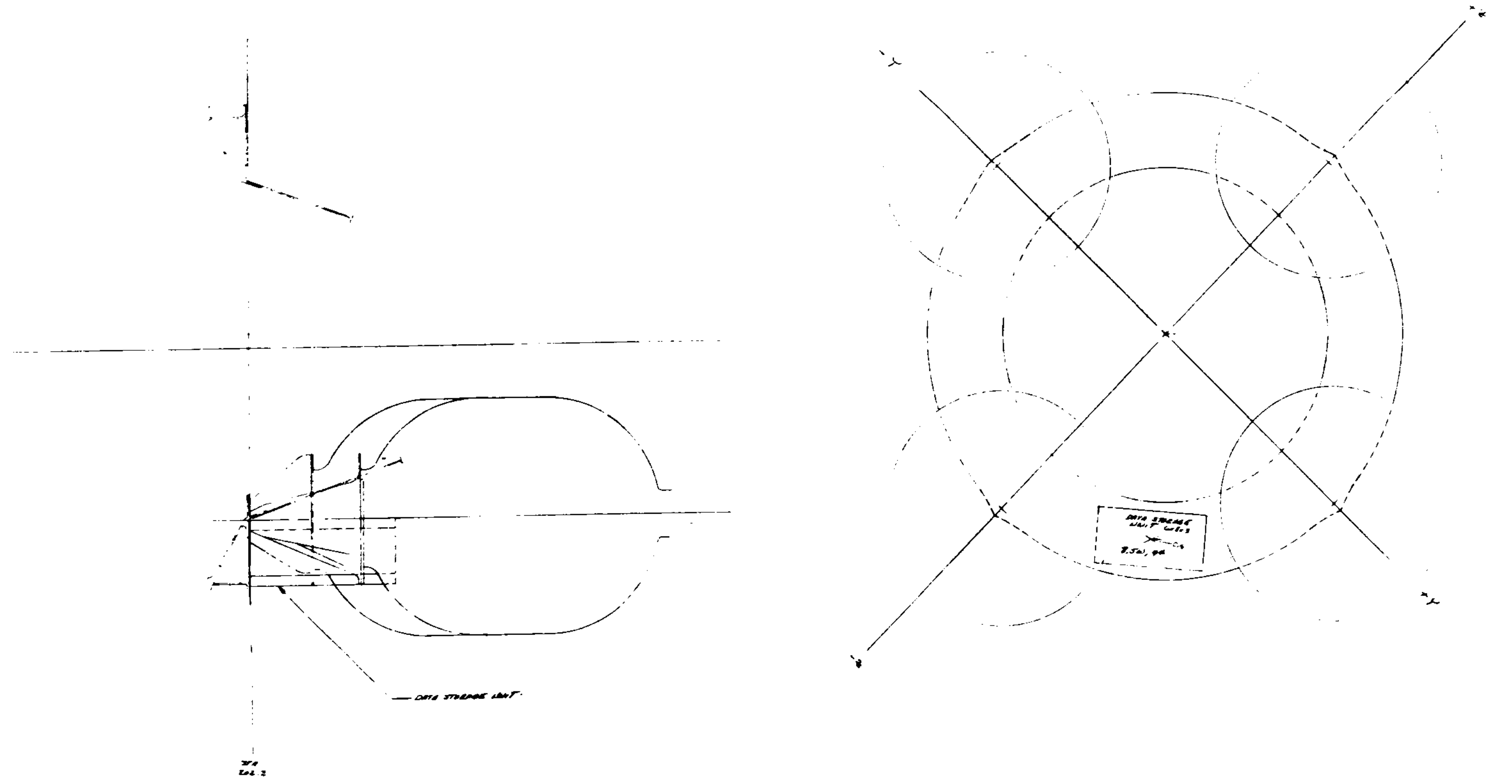

FIGURE 4.0.3.13

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LUNAR ORBITER ADAPTABILITY STUDY CASE I

LO-JWR-2

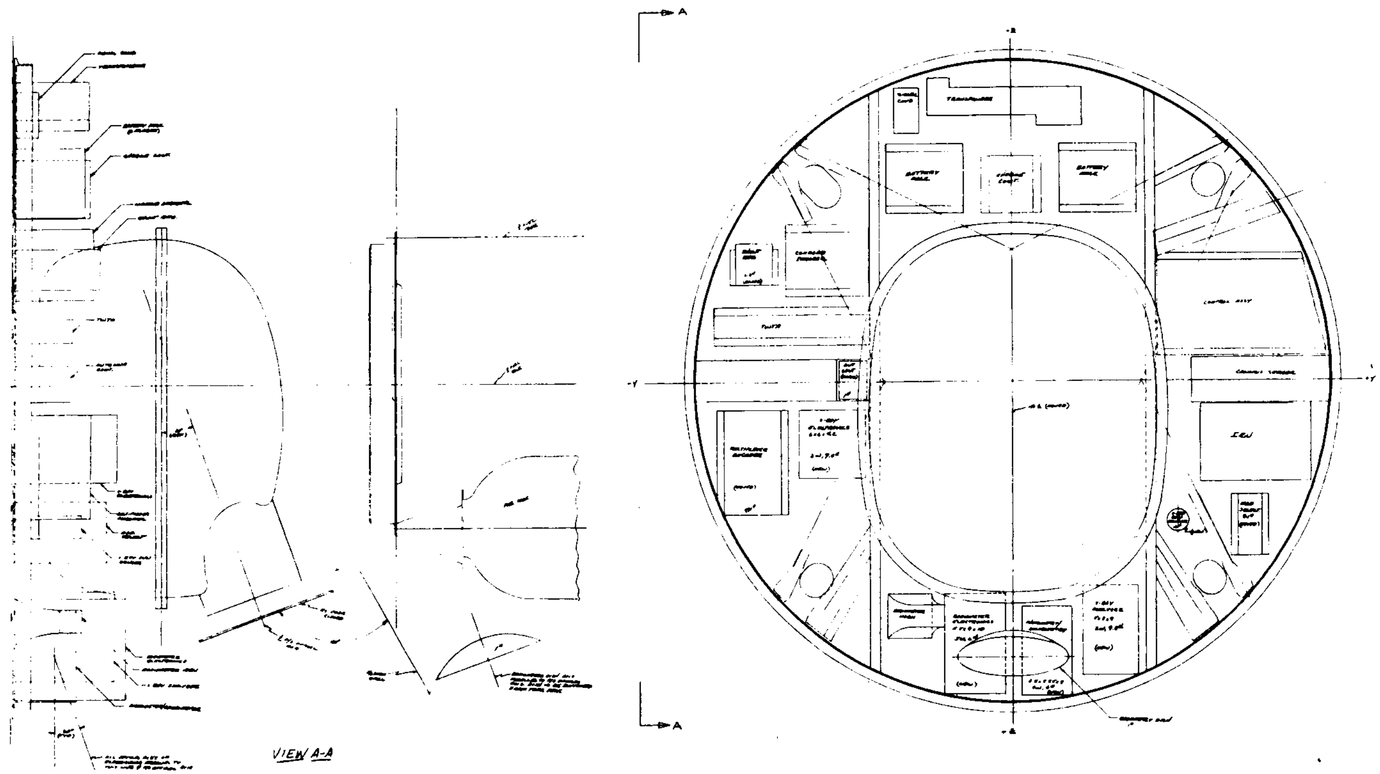

FIGURE 4.0.3.15

BOEING NO D2-100369-1

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LUNAR ORBITER ADAPTABILITY STUDY CASE IIc

LO-MR-5

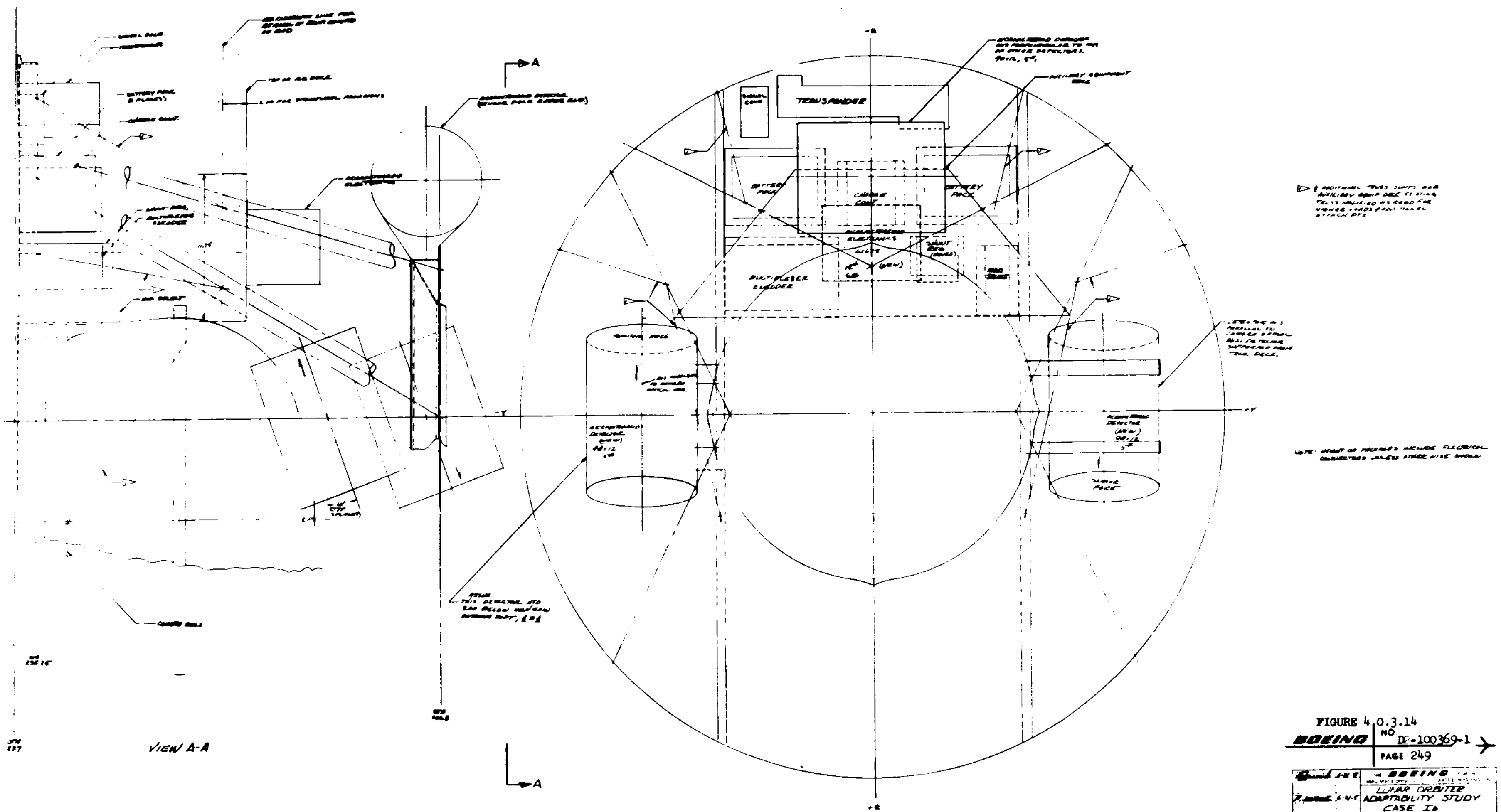

FIGURE 4.0.3.14

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LUNAR ORBITER ADAPTABILITY STUDY CASE Ib
L.O-JWR-2

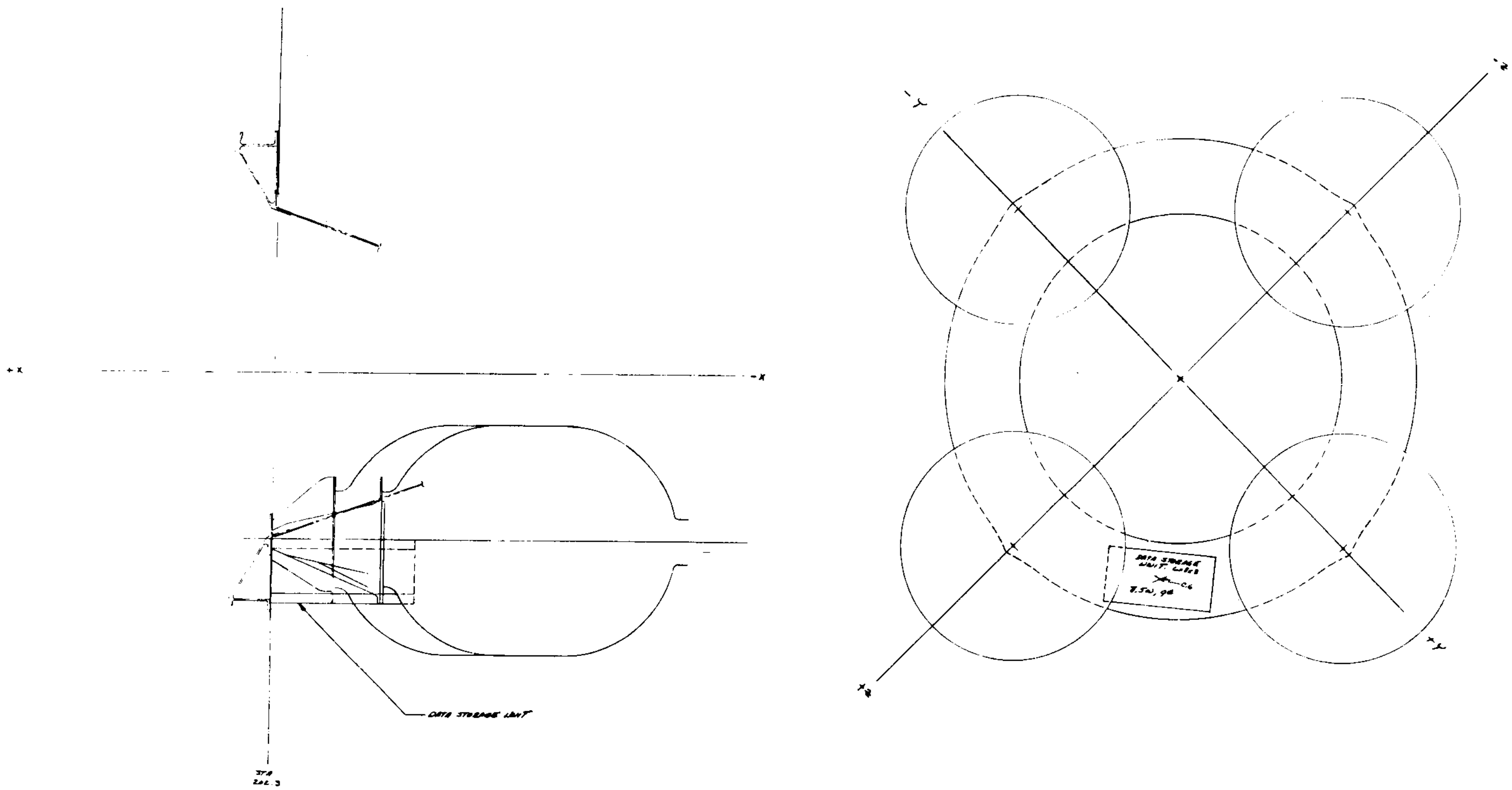

FIGURE 4.0.3.16

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LUNAR ORBITER
ADAPTABILITY STUDY
CASE IIa

LO-JWR-5

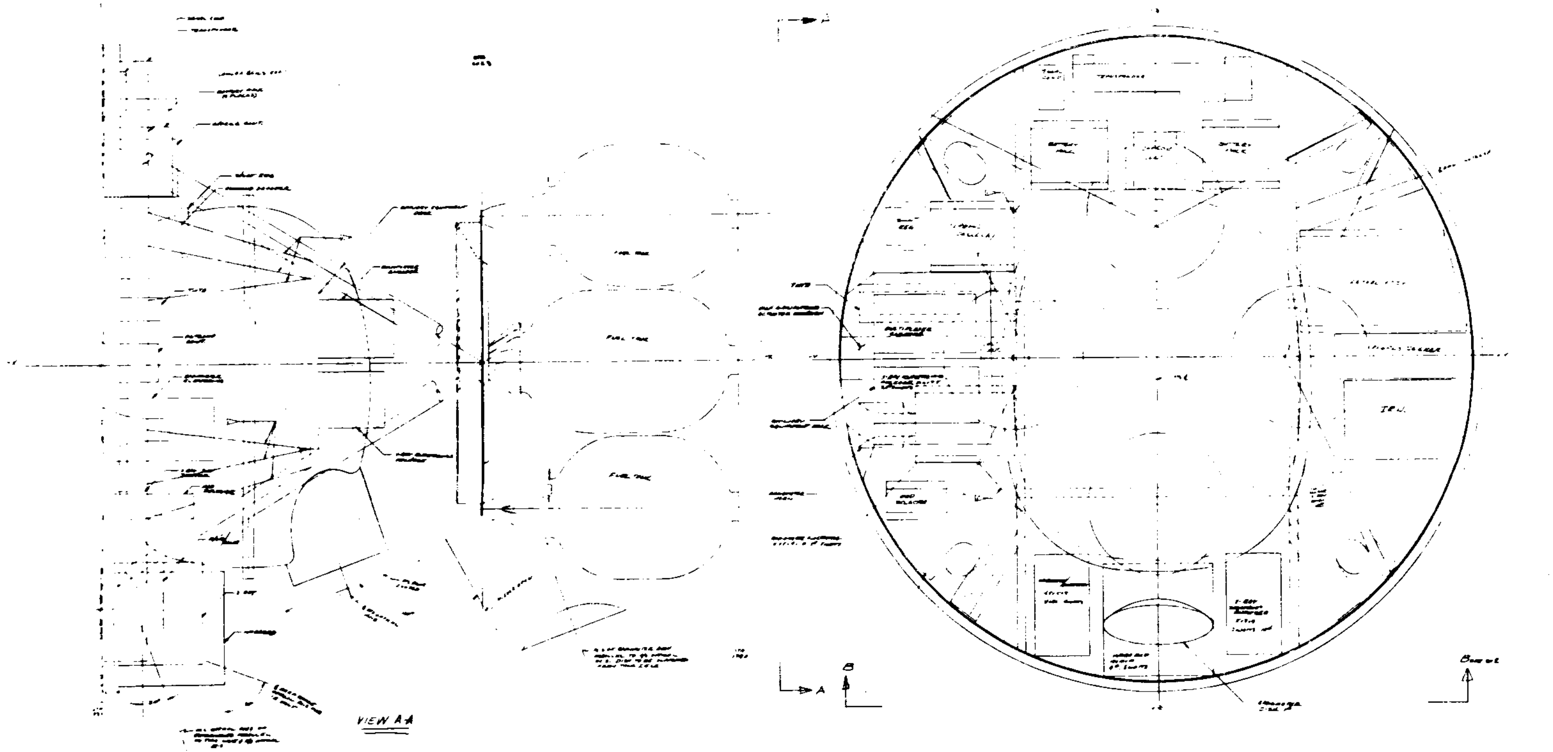

FIGURE 4.0.3.17

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LUNAR ORBITER
ADAPTABILITY STUDY
CASE II

LO-JWR-3

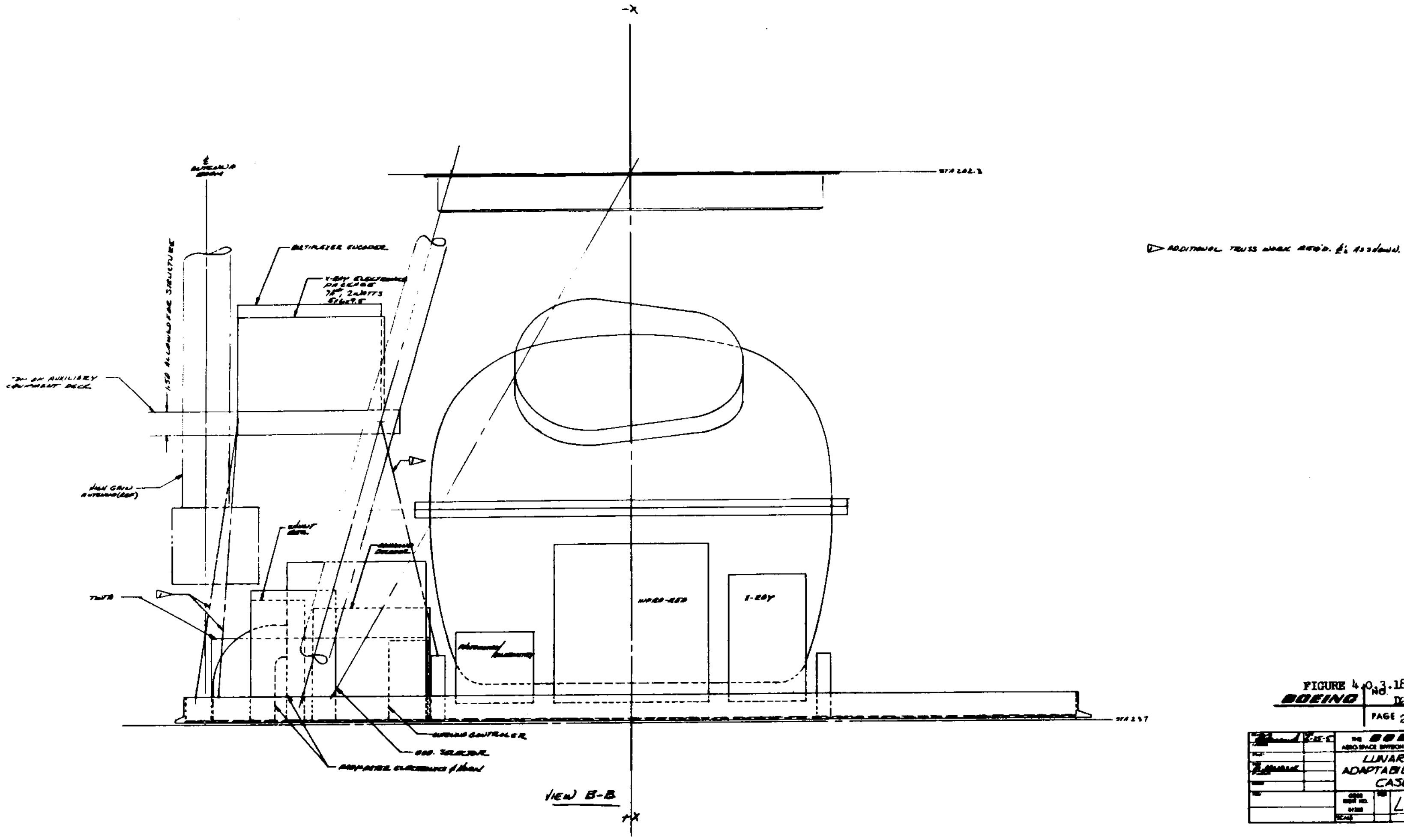

FIGURE 4.0.3.18

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LUNAR ORBITER ADAPTABILITY STUDY CASE IIc

LO-JWR-3

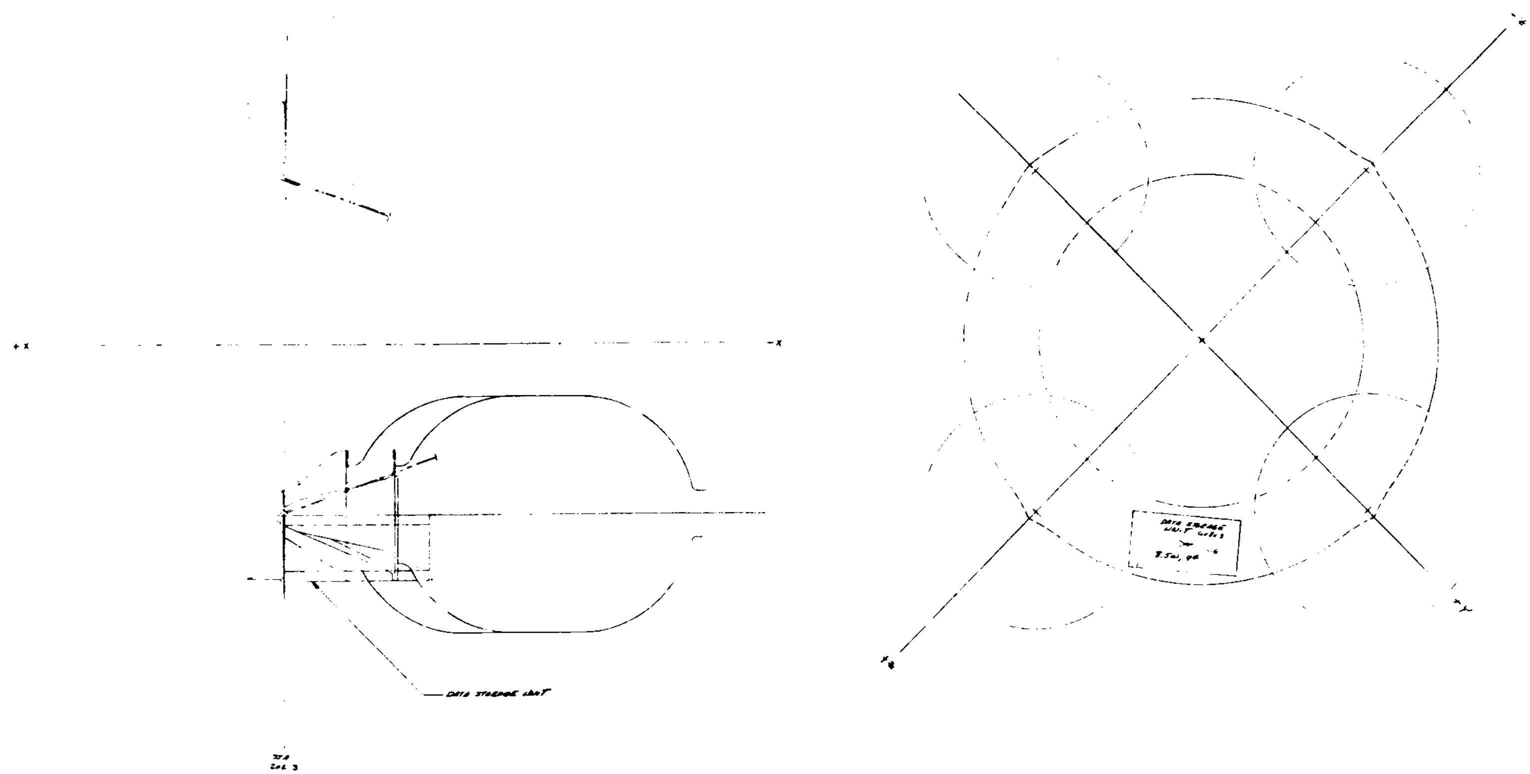

FIGURE 4.0.3.19
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LUNAR ORBITER
ADAPTABILITY STUDY
CASE IIc
LO-JWR-3

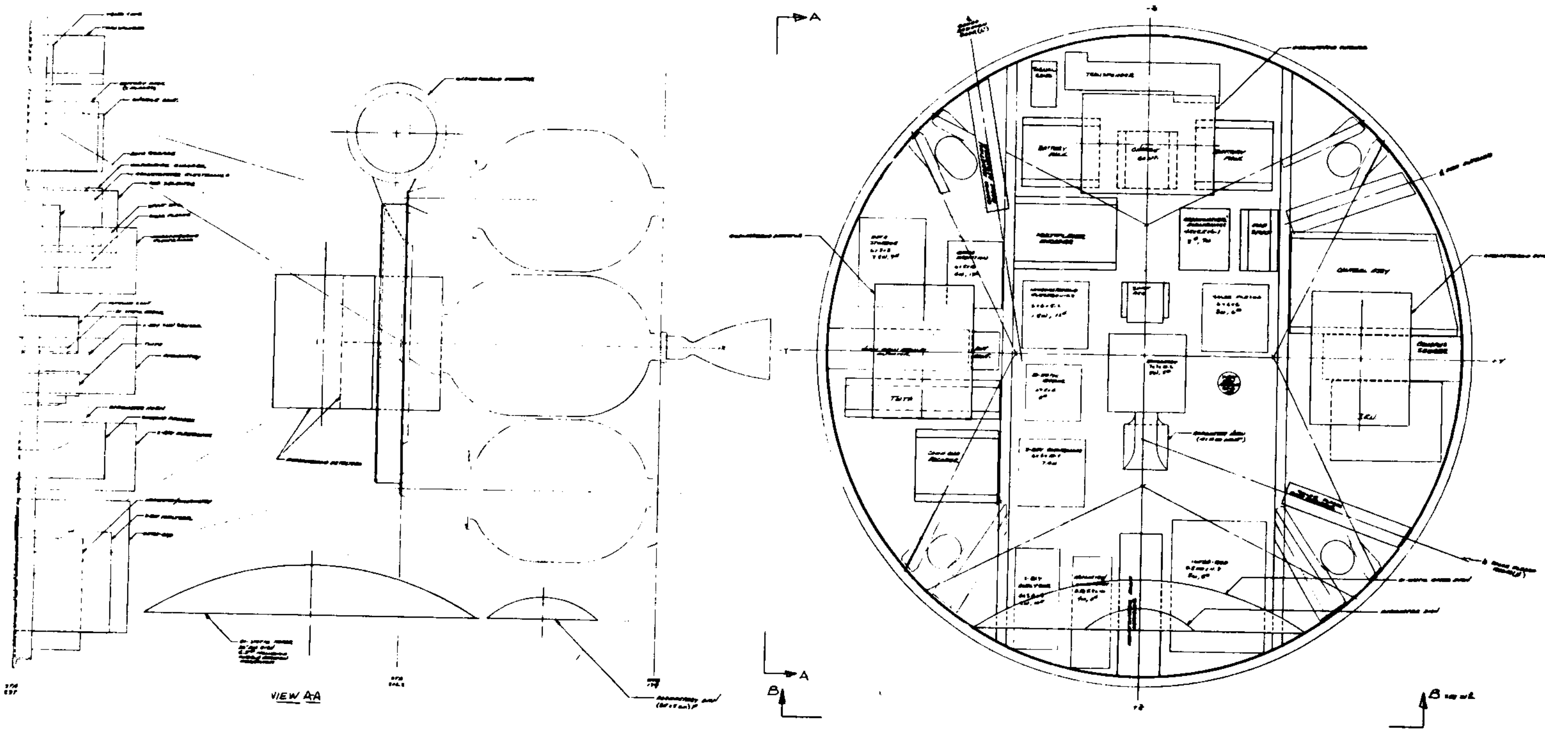

FIGURE 4.0.3.20

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LUNAR ORBITER ADAPTABILITY STUDY CASE III

LO-JWR-4

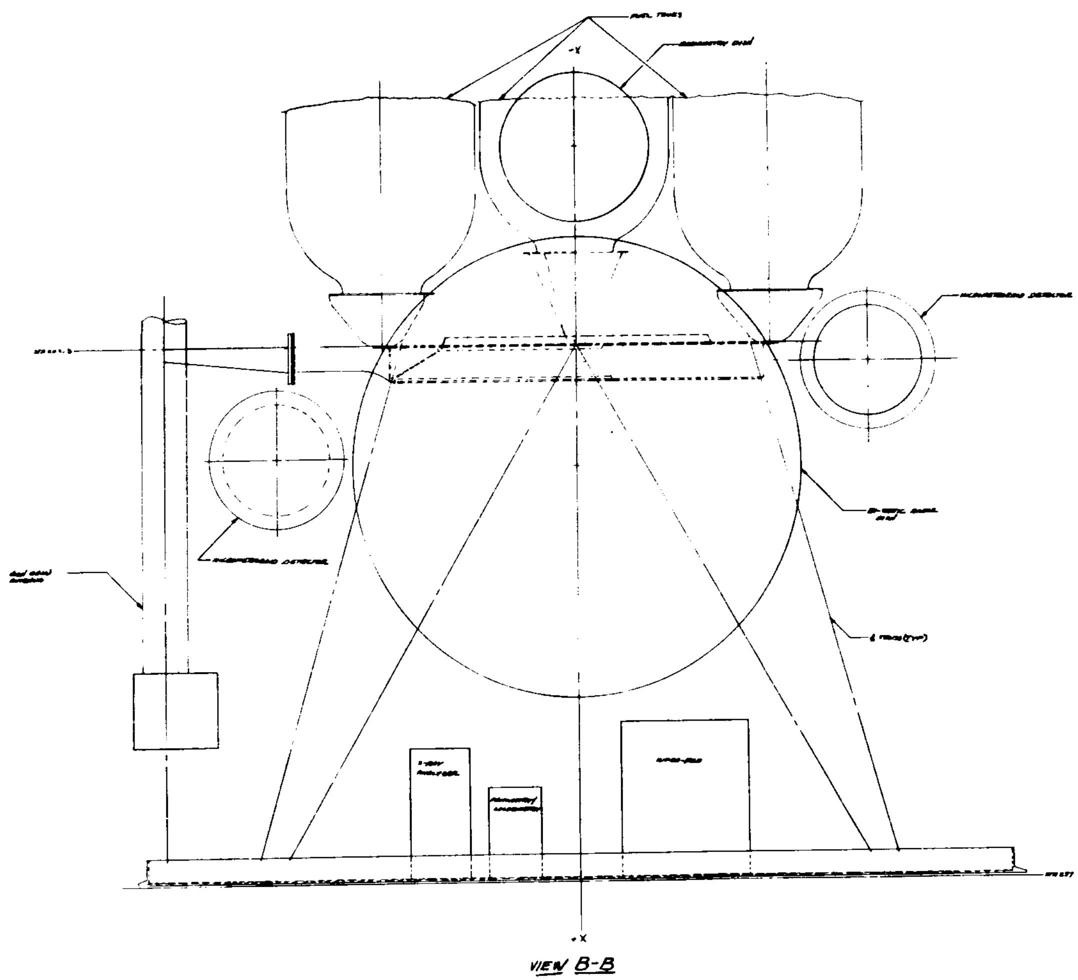

FIGURE 4.0.3.21

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LUNAR ORBITER ADAPTABILITY STUDY-CASE III

LO-JWR-4